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# DESIGN CHARACTERISTICS AND PERFORMANCE OF A COMBUSTOR FOR USE AS AN IONIZED GAS SOURCE IN MAGNETOHYDRODYNAMIC POWER GENERATION STUDIES

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ARO, Inc.

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# NOTICES

# DESIGN CHARACTERISTICS AND PERFORMANCE OF A COMBUSTOR FOR USE AS AN IONIZED GAS SOURCE IN MAGNETOHYDRODYNAMIC POWER GENERATION STUDIES

C. R. Darlington, R. E. Gilburth, and R. S. Ballard
ARO, Inc.

## FOREWORD

The test program reported herein was conducted at the request of the Air Force Aero-Propulsion Laboratory (AFAPL), Air Force Systems Command (AFSC), Wright-Patterson Air Force Base, Ohio, under Contract AF33(615)-2691 for the Chrysler Corporation, Space Division, Huntsville Operations, under Program Element 62402F, Project 3145.

The results of the test were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), AFSC, Arnold Air Force Station, Tennessee, under Contract F40600-69-C-0001. The test was conducted in Propulsion Research Area (R-2C-4) of the Rocket Test Facility (RTF) from April 26, 1968, to March 27, 1969, under ARO Project Number RW0903, and the manuscript was submitted for publication on July 2, 1969.

Information in this report is embargoed under the Department of State International Traffic in Arms Regulations. This report may be released to foreign governments by departments or agencies of the U. S. Government subject to approval of the Air Force Aero-Propulsion Laboratory (APIE-2), or higher authority within the Department of the Air Force. Private individuals or firms require a Department of State export license.

This technical report has been reviewed and is approved.

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## ABSTRACT

Design characteristics and performance of a combustor for use as a high energy, ionized gas source in magnetohydrodynamic power generator studies are described. The liquid oxygen ($LO_2$)/JP-4 combustor was operated over a chamber pressure range from 240 to 300 psia and at a characteristic exhaust velocity efficiency of 91 ± 1 percent for oxidizer/fuel ratios ranging from 2.0 to 3.1. Combustor power output was approximately 17.5 to 20.5 megawatts (MW) over its range of operation. Provisions were incorporated into the design for injection of a saturated solution of water and cesium carbonate seeding agent into the thrust chamber to provide a high ion concentration in the exhaust gases.

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and oxidizer-to-fuel flow ratio (O/F) with the exhaust gases expanding to a specific value of nozzle exit static pressure.

The design criteria for the hot ionized gas source (combustor) described herein departs from traditional design considerations, which are normally concerned with obtaining a maximum specific impulse at a specific operating condition, in that the combustor discussed herein was required to operate over a wide range of O/F ratios and chamber pressures. The final design was based on experience gained in developing small thrust rocket engines used in previous test programs in RTF and on information contained in published literature.

The primary function of the combustor is to provide a high energy (approximately 20 MW of power), ionized gas source for application in an MHD power generation system. The combustor design is based on the following specified design criteria:

1. Propellants: $LO_2$/JP-4
2. Propellant Total Mass Flow Rate: 3.75 to 4.25 $lb_m$/sec
3. Oxidizer/Fuel Flow Ratio: 2.0 to 2.8
4. Provisions for Seed Injection into Combustion Chamber
5. Exhaust Nozzle Exit Half-Angle: 0 deg
6. Exhaust Nozzle Exit Diameter: 2.0 in.
7. Nozzle Throat Diameter: 1.76 in.
8. Run Duration: To 20 min

In addition to the above design criteria, it was required that the combustor interface with the MHD channels and provide a sufficient annulus clearance between the MHD channel support tube and combustor for MHD channel electrical and instrumentation cables and MHD channel coolant lines. Consequently, the diameter of the combustor, including the cooling (water) jacket was limited to 4.75 in.

In the following sections, the design requirements and characteristics of the injector, the combustion chamber, and the nozzle are described. Details of the combustor coolant system are discussed. A schematic of the combustor configuration is presented in Fig. 1 (Appendix I). The resultant combustor design data are presented in Table I (Appendix II).

## 2.2 INJECTOR DESIGN

The design of an injector is largely empirical since a complete theory for relating design parameters to combustor performance and combustion phenomena has not been devised (Ref. 3).

The primary functions of the injector are to introduce and meter the propellant flow into the combustion chamber and to atomize and mix the propellants in such a manner that a correctly proportioned, homogeneous fuel-oxidizer mixture will result that can be readily vaporized and burned.

Coaxial-type injectors for 500- and 1000-$lb_f$ thrust $LO_2$/RP-1 combustors developed during previous RTF test programs have proved to be successful in performing these functions. Therefore, a coaxial-type injector design approach was used for the combustor (Fig. 2).

The injector for an $LO_2$/JP-4 combustor would normally be designed to operate at an O/F ratio of approximately 2.2 to achieve maximum specific impulse. However, when the requirement exists for the combustor to operate at optimum efficiency over a range of O/F ratios from 2.0 to 2.8, experience gained in previous coaxial-type injector development programs indicated that optimum performance is achieved by using a higher design O/F ratio. Therefore, an injector design O/F ratio of 2.6 was chosen. Criteria used to size the oxidizer/fuel set flow areas were based on an O/F velocity ratio of 2.0 while maintaining both oxidizer and fuel injector pressure drop within 10 to 50 percent of chamber pressure.

The coaxial-type injector (Fig. 2b) selected for the combustor has 54 oxidizer tube-fuel annuli sets located in three concentric rings. Liquid oxygen and JP-4 fuel were injected into the combustion chamber through tubes and annuli, respectively. This arrangement (oxidizer tube-fuel annuli) was used to promote atomization and mixing, to reduce injector face heating, and to provide fuel cooling for the thrust chamber walls.

The injector oxidizer/fuel sets were sized to meet the velocity ratio-pressure drop design criteria. Fuel annuli were 0.073-in.-OD and 0.064-in.-ID. The oxidizer tubes (0.064-in.-OD and 0.045-in.-ID) were crimped (Fig. 2b detail) to a minimum width of 0.021 in. to meet the velocity-pressure drop criteria, to increase atomization, and to improve fuel-oxidizer mixing. The oxidizer tubes were extended 0.075 in. from the injector face to prevent excessive injector face heating and to promote mixing of the oxidizer and fuel in the vaporized state.

All parts of the injector were fabricated from type 347 stainless steel. Photographs of the injector are shown in Figs. 2c and d.

The injector is equipped with a removable igniter fuel-seed pin (Fig. 3) positioned in the center of the injector face. The pin is designed to accomplish four individual functions:

1. Provide a passage for injection of a pyrophoric igniter fuel (triethylborane or TEB) into the combustion chamber during the ignition phase,
2. Provide a passage for injection of secondary fuel (JP-4) into the combustion chamber after the ignition phase,
3. Provide a passage for injection of seed (saturated solution of water and $Cs_2CO_3$) into the combustion chamber, and
4. Provide a pressure port for measuring combustion chamber pressure.

The igniter fuel (TEB) was injected into the combustion chamber through six V-shaped grooves at an angle to induce a swirling action which increases the mixing of the igniter with the oxidizer. After ignition, secondary fuel (JP-4) was injected into the combustion chamber through the igniter grooves. Control of the secondary fuel flow rate provides a means for controlling the main fuel flow injection velocity. During previous developmental testing of a similar combustor configuration, optimum combustor performance was obtained with a secondary fuel flow rate of approximately 30 percent of the total fuel flow. Therefore, a secondary fuel flow rate of 30 ± 2 percent of the total fuel flow rate was used during all combustor firings during this test program.

A seeding material consisting of 70-percent (by weight) solution of $Cs_2CO_3$ in water was injected into the combustion chamber through six orifices in the igniter fuel-seed pin (Fig. 3) to provide a high ion concentration in the exhaust gases. The orifices were sized to inject the seeding agent into the thrust chamber at controlled flow rates consistent with available supply pressures. The individual seed injection system was used to avoid fluid and propellant valving incompatibility problems which could be encountered by injecting the seed into the combustion chamber through the main fuel annuli or igniter-secondary fuel grooves. An individual seed injection system also increases the versatility of the injector since different types of seeding agents may be used without affecting the operation of the igniter, fuel, or oxidizer systems.

In addition to performing the above functions, the igniter fuel-seed pin was equipped with a port for measuring steady-state chamber pressure.

# SECTION I
# INTRODUCTION

A magnetohydrodynamic (MHD) electric power generator is classed as a direct-energy conversion device. Ionized gas flowing at high velocity through a channel is acted upon by a transverse magnetic field to produce an electromotive force perpendicular to the velocity vector and the magnetic field. The same physical principles are involved in an MHD generator as in a conventional electric generator except that conducting gases replace the current-carrying conductors of the rotor.

Chrysler Corporation, Huntsville Operations, is currently engaged in a research and development program aimed at the development of a 1-MW, flightweight, MHD generator system to power a plasma arc illuminator (Refs. 1 and 2). Primary components of the system will include a combustor (plasma generator) coupled to an MHD channel immersed in a magnetic field provided by a superconducting magnet. The output of the MHD generator will power the plasma arc illuminator.

Part of the overall development program was the requirement to design, fabricate, and qualify a liquid oxygen ($LO_2$)/JP-4 combustor to provide the high energy (approximately 20 MW) ionized gases for the MHD channel. The design and fabrication of the MHD combustor and its associated systems were performed by ARO, Inc., personnel. Testing of the MHD combustor was conducted in Propulsion Research Area (R-2C-4) of the Rocket Test Facility (RTF).

The design concepts and operating characteristics of the $LO_2$/JP-4 MHD combustor are presented herein. Performance characteristics both with and without injection of a saturated water and cesium carbonate ($Cs_2CO_3$) solution into the thrust chamber are discussed.

# SECTION II
# COMBUSTOR DESIGN CHARACTERISTICS

## 2.1 DESIGN REQUIREMENTS

Design criteria for small thrust ($\approx$1000 $lb_f$), liquid-propellant rocket engines are not well defined in the literature since each particular design depends on the application. Normally, a combustor consisting of an injector, combustion chamber, and nozzle is designed to operate with a given propellant combination at a constant combustion chamber pressure

## 2.3 THRUST CHAMBER DESIGN

The thrust chamber (combustion chamber and nozzle, Figs. 4a and b) was designed to provide adequate mixing and combustion of the propellants and to provide uniform, parallel flow at the nozzle exit. The thrust chamber was fabricated from Mallory® 3 copper alloy because of the excellent thermal and physical properties of the material.

The inside diameter of the combustion chamber (Fig. 4a) was 3.75-in. Combustion contraction ratio (chamber cross-sectional area/throat area) was 3.61 and was designed to avoid a substantial loss in energy efficiency (Ref. 3). A combustion chamber-to-nozzle throat convergence half-angle of 30 deg was used to ensure a steady combustion gas velocity increase and a uniform combustion gas velocity profile as the combustion gas approached the nozzle throat (Ref. 4). Combustion chamber length (injector face-to-nozzle throat) was 10.86 in., and the combustor characteristic length (L*) was approximately 33 in.

The nozzle is defined as the part of the thrust chamber from the throat to the exit of the thrust chamber. Coordinates (Fig. 4a) for an axially symmetric supersonic wind tunnel test nozzle designed by the method of characteristics for inviscid flow of a perfect gas (Refs. 5 and 6) were used to establish the combustor nozzle geometry. A method of characteristics computer program for a perfect gas with a specific heat ratio ($\gamma$) of 1.22 (Ref. 7) was used to determine the Mach number profile (Fig. 5) and flow angularity (Fig. 6) at the nozzle exit to further substantiate the design coordinates. Nozzle exit Mach number was computed to be $1.57 \pm 0.01$. A boundary-layer computer program (Ref. 8) was used to determine the boundary layer in the nozzle. Boundary-layer thickness was computed to be approximately 0.0001 in. at the nozzle exit.

## 2.4 COMBUSTOR COLLANT SYSTEM

Run durations up to 20 min, specified in the design criteria (Section 2.1), required cooling the combustor to ensure successful, safe operation. The combustion chamber and nozzle were enclosed by a water jacket to provide cooling for the assembly (Fig. 1).

Generally, small thrust (<1000 $lb_f$), work-horse-type combustors can be successfully cooled with a water flow rate 1.5 to 3.0 times the total propellant mass flow rate and with a water velocity at the nozzle throat of 80 to 100 ft/sec. A coolant velocity of one-half the required throat velocity is usually adequate to cool the combustion chamber walls and the divergent walls of the nozzle.

The coolant criteria (flow rate and velocity) and a heat transfer computer program (based on equations in Ref. 3) developed during previous combustor development programs at RTF were used in an iterative-type process to establish the combustor coolant system requirements. The heat transfer program (RTF Program No. 57) calculated the combustor heat transfer rates and temperatures for an assumed water temperature increase (outlet water temperature minus inlet water temperature) in increments of 10°F. Primary inputs to the computer program, in addition to the water velocity and flow rate, were the thrust chamber thermal conductivity and physical dimensions and the combustor gas properties (conductivity, specific heats, temperature, viscosity, and velocity).

Cooling water flow was supplied to the inlet at a pressure of 200 psia and at a flow rate of 15 $lb_m$/sec. Outlet water pressure was maintained at 100 psia to prevent the possibility of cavitation and to ensure that the water pressure exceeded the critical pressure level. Thrust chamber annular water passages were sized to provide a water velocity of 45 and 90 ft/sec (for a flow rate of 15 $lb_m$/sec) along the combustion chamber and nozzle throat walls, respectively.

Typical calculated heat transfer results used to establish the validity of the assumed cooling water flow rate and velocity are shown in Figs. 7a, b, and c for the combustion chamber, nozzle throat, and nozzle exit walls, respectively. The combustor coolant system has proved to be successful.

## SECTION III
## INSTALLATION AND TEST PROCEDURE

### 3.1 INSTALLATION

The combustor was installed in Propulsion Research Area (R-2C-4) of RTF to determine the operating characteristics and performance of the combustor (Fig. 8). The combustor was mounted on a support system and connected to the facility propellant and coolant systems. The combustor nozzle exit flange was aligned with and bolted to the exhaust diffuser flange. The 12-in.-diam, 8-ft-long exhaust diffuser extended through a rubber slip-joint seal at the forward bulkhead of a spray chamber, containing one air spray ring and six water spray rings. A 12-in. exhaust duct was bolted to the downstream end of the spray chamber to direct the cooled exhaust gases into the facility exhaust ducting to be discharged into the atmosphere.

A schematic of the propellant system is shown in Fig. 9. Combustor ignition was accomplished with 0.5 $lb_m$ of TEB pyrophoric fuel. The $LO_2$ was supplied from two 550-gal. tanks pressurized with gaseous nitrogen ($GN_2$). An automatic pressure control system maintained tank pressure during firing at a value that provided the desired flow rate.

The JP-4 fuel was supplied to an aircraft-type fuel pump from facility storage at a pressure of 60 psia. The desired combustor JP-4 flow rate was provided by adjustment of a fuel bypass system back to the facility fuel storage reservoir.

The $Cs_2CO_3$ seeding agent was supplied to the combustor from a seed-charged cylinder. The seeding agent was discharged from the cylinder by a piston driven with water pressurized with $GN_2$. The water was stored in a 75-gal. tank. All propellant systems incorporated provisions for purging the lines with dry $GN_2$.

### 3.2 INSTRUMENTATION

Instrumentation (Table II and Fig. 9) was provided to measure the combustor chamber pressure, injector pressures, propellant and seed tank pressures, nozzle exit pressures, propellant and seed flow rates, and combustor cooling flow rate and temperature rise.

Bonded strain-gage-type transducers were used to measure pressures. Propellant, seed, and cooling water flow rates were measured with turbine-type flowmeters. Iron-constantan thermocouples were used to measure fuel temperatures. Copper-constantan thermocouples were used to measure cooling water, $LO_2$, and seed temperatures.

Combustor pressure, flow, and temperature data were recorded on magnetic tape from a multi-input high-speed, analog-to-digital converter at a scan rate for each channel of 75 times/sec. A photographically recording, galvanometer-type oscillograph and null-balance potentiometer-type setup chart recorders provided an independent backup of selected instrumentation channels and a means for monitoring test activity.

### 3.3 CALIBRATION

All transducers and system calibrations (Table II) performed during this test program are traceable to the National Bureau of Standards (NBS).

Each link in the traceability chain back to the NBS is maintained and documented by the AEDC Standards Laboratory (Ref. 9).

The flow measurement transducers, installed in a section of their associated piping, were calibrated in the Standards Laboratory utilizing a dynamic weigh-type water flow calibration system to determine their volumetric flow versus frequency output relationship. Before and after each test period, the flow recording systems were calibrated by applying known frequency input levels from a frequency generator calibrated in the Standards Laboratory.

The temperature transducers (thermocouples) were fabricated from wire conforming to Instrument Society of America specifications. Before and after each test period, known millivolt levels were applied to each temperature recording system, and the corresponding temperature equivalents were obtained from 150°F reference tables based on the NBS temperature versus millivolt tables. Nonlinearity in the thermocouple characteristics were accounted for in the data reduction program.

The pressure measuring transducers were calibrated in the Standards Laboratory to establish their applied pressure versus resistance shunt equivalent pressure relationship. Before and after each test period, multiple-step resistance shunt calibrations were performed to calibrate the pressure recording system.

### 3.4 TEST PROCEDURE

Before each test period the instrumentation systems were calibrated, and the propellant, seed, and coolant systems were prepared for operation. When the pressures were set to give the desired combustor flow rates, a combustor firing sequence was initiated at a time designated as $T_0$ which automatically controlled the events as shown typically in Figs. 10 and 11 and time sequenced as follows:

| | |
|---|---|
| $T_0$ | Automatic firing sequence started. |
| $T_0$ + 0.6 sec | Igniter (TEB) valve opened (TEB charge depleted in approximately 1 sec and followed by JP-4). |
| $T_0$ + 0.8 sec | $LO_2$ propellant valve opened. |
| $T_0$ + 0.95 sec | $LO_2$-TEB ignition. |
| $T_0$ + 1.00 sec | $LO_2$-TEB combustion chamber pressure switch satisfied (75 psia), signal for fuel valve to open. |

| | |
|---|---|
| $T_0$ + 1.50 sec | Fuel valve opened. |
| $T_0$ + 1.70 sec | Main stage steady-state combustion chamber pressure established. |
| $T_0$ + 2.00 sec | Seed valve opened. |

The engine firing duration was nominally 5 to 20 sec, at which time the firing was terminated by the automatic firing sequence normal combustor shutdown at a time designated as $T_s$. Typical shutdown events occurred as follows:

| | |
|---|---|
| $T_s$ - 0.5 sec | Seed valve closed. |
| $T_s$ | Normal shutdown event initiated upon elapse of prescribed run time. |
| $T_s$ + 0.22 sec | $LO_2$ propellant valve closed. |
| $T_s$ + 0.27 sec | Igniter valve closed. |
| $T_s$ + 0.46 sec | Fuel valve closed. |

When the engine firing was completed, the injector was purged with $GN_2$ to remove contaminants from the thrust chamber.

# SECTION IV
# COMBUSTOR OPERATING CHARACTERISTICS AND PERFORMANCE

The objective of the test program reported herein was to design, fabricate, and qualify an $LO_2$/JP-4 combustor to provide a hot ionized gas source for use in an MHD power generator system.

In the sections to follow, the operating characteristics of the combustor components (injector, combustion chamber, and nozzle) and the performance of the combustor both with and without seed injection are discussed.

## 4.1 INJECTOR

Injector oxidizer and fuel velocities were approximately 95 and 45 ft/sec, respectively, corresponding to an oxidizer and main fuel injector pressure drop of 110 and 35 psid for mass flow rates of 2.9 and 0.8 $lb_m$/sec, respectively.

Injector pressure drop variations with flow rates are shown in Fig. 12 for $LO_2$, main JP-4 fuel, secondary JP-4 fuel, and a saturated solution of water and $Cs_2CO_3$.

During the initial combustor checkout test, the fuel annuli became clogged by the $LO_2$-igniter (TEB) combustion products during the ignition sequence. Clogging was subsequently prevented by increasing the fuel annuli $GN_2$ purge pressure to a value slightly exceeding (5 to 20 psi) the $LO_2$-TEB combustion chamber pressure.

The igniter fuel (TEB) injected into the thrust chamber was followed by JP-4 fuel (secondary JP-4 fuel) through the igniter ports. Approximately 30 percent of the total JP-4 fuel was injected into the chamber through the igniter ports to improve fuel-oxidizer mixing and to provide cooling for the injector face.

## 4.2 COMBUSTION CHAMBER AND NOZZLE EXIT PRESSURES

The combustor exhibited a smooth starting characteristic and operated with a stable combustion chamber pressure during all combustor firings. Combustion chamber pressure fluctuation was nominally ±5 psia for all operating conditions (Fig. 11). Seed solution injected into the thrust did not affect the combustion chamber pressure stability.

Combustor nozzle exit static pressure measurements were made 0.2 and 0..4 in. downstream of the exit plane in a 2.00-in.-diam section attached to the combustor. The ratio of exit static pressure to chamber pressure ranged from 0.30 to 0.34, compared with a theoretical pressure ratio of 0.265 for frozen composition (Ref. 10) and with a theoretical pressure ratio of 0.29 for equilibrium composition (Ref. 11).

## 4.3 CHARACTERISTIC EXHAUST VELOCITY VARIATION WITH O/F RATIO

The characteristic exhaust velocity (c*) of a combustor is frequently used as a measure of the energy available after combustion and can be expressed as

$$c^* = \frac{P_c A_t g_c}{\dot{w}_t}$$

where $P_c$ is the combustion chamber pressure ($lb_f/in.^2$) measured at the injector face, $A_t$ is the nozzle throat area ($in.^2$), $g_c$ is a dimensional constant (32.2 $lb_m$-ft/$lb_f$-$sec^2$), and $\dot{w}_t$ is the total propellant flow rate ($lb_m$/sec). The combustor efficiency may be defined as the ratio of the

measured c* to theoretical c*. The theoretical c* used in this report is based on frozen composition (Ref. 10).

The combustor was operated at a chamber pressure ranging from 240 to 300 psia for O/F ratios ranging from 2.0 to 3.1. The variation of c* with O/F ratio is shown in Fig. 13. The value of c* ranged from a minimum of 5010 ft/sec at an O/F ratio of 3.1 to a peak of 5220 ft/sec at an O/F ratio of 2.2. Engine efficiency was nominally 91 ± 1.0 percent for O/F ratios ranging from 2.0 to 3.1 (Fig. 14). Engine efficiency was approximately 3 percentage points less than the efficiency used to establish the design criteria. Combustor power output was approximately 17.5 to 20.5 MW.

## 4.4 COMBUSTOR OPERATION WITH SEED FLOW

The seed material used during this investigation was a 70-percent (by weight) solution of $Cs_2CO_3$ in water. The percent seed flow is defined as

$$[\dot{w}_s/(\dot{w}_t + \dot{w}_s)] \times 100$$

where $\dot{w}_s$ is the seed flow rate ($lb_m$/sec). Arbitrarily, the combustor characteristic exhaust velocity with seed flow ($c_s$*) is defined as

$$c_s^* = \frac{P_c A_t g_c}{\dot{w}_t + \dot{w}_s}$$

Seed was injected into the combustion chamber approximately 1.0 sec after combustor ignition and was continued until approximately 0.5 sec before combustor shutdown. The seed flow rate was varied from 0 to 0.64 $lb_m$/sec at a nominal $LO_2$/JP-4 mixture ratio of 2.86 and at a total propellant ($LO_2$ + JP-4) weight flow of 4.0 $lb_m$/sec.

The effect of percent seed flow rate on $c_s$* is shown in Fig. 15. The $c_s$* decreased from 5100 ft/sec at a seed flow rate of 0 percent to 4550 ft/sec at a seed flow rate of 14 percent. The low values of $c_s$* are attributed to the noncombustible seed solution being injected into the combustion chamber.

## 4.5 COMBUSTOR HEAT TRANSFER RATES

Heat transfer rates (computer-calculated) for the combustion chamber, nozzle throat, and nozzle exit are shown extrapolated for the axial length of the thrust chamber in Fig. 16. The heat transfer rates and the applicable thrust chamber wall surface area were used to calculate

the heat transferred from the thrust chamber to the coolant. The thrust chamber heat transfer calculated in this manner was approximately 380 Btu/sec.

During the combustor development firings, the coolant water flow rate and the coolant water temperature increase (outlet water temperature minus inlet water temperature) were measured. Measured thrust chamber heat transfer, calculated by the equation

$$\dot{q} = \dot{w}_c \bar{c}_p \Delta T$$

where

$q$ = Thrust chamber heat transfer, Btu/sec

$\dot{w}_c$ = Coolant water flow rate, $lb_m$/sec

$\bar{c}_p$ = Average specific heat of water at 80°F, Btu/°F $lb_m$

$\Delta T$ = Outlet-inlet coolant water temperature, °F

was 300 ± 30 Btu/sec, as compared with a computed value of 380 Btu/sec. The difference between measured and computed heat transfer values is primarily attributed to the difference in the assumed c* of 5350 ft/sec and the c* ≈ of 5200 ft/sec generally encountered during testing.

## SECTION V
## SUMMARY

A liquid oxygen ($LO_2$)/JP-4 combustor was designed to specific criteria and fabricated for use as a hot ionized gas source in magnetohydrodynamic (MHD) power generator system. A water solution of cesium carbonate was injected into the combustion chamber to produce a high ion concentration in the high velocity exhaust gases. The combustor operating characteristics are summarized as follows:

1. The combustor operated successfully at oxidizer-to-fuel (O/F) ratios ranging from 2.0 to 3.1, for combustion chamber pressures ranging from 240 to 300 psia, and for seed flow rates ranging from 0 to 0.64 $lb_m$/sec. Combustor power output ranged from approximately 17.5 to 20.5 MW.

2. The combustor characteristic exhaust velocity (c*) without seed flow ranged from 5220 to 5010 for O/F ratios ranging from 2.0 to 3.1. The c* efficiency was 91 ± 1.0 percent over the range of combustor operation.

3. At an O/F ratio of 2.86, the characteristic exhaust velocity ($c_s$*) of the combustor decreased from 5100 to 4550 ft/sec when the seed flow rate was increased from 0 to 14 percent of the total flow rate (propellant and seed).

4. Pressure measurements obtained during combustor firings indicated that the nozzle exit static/total pressure ratio was $0.32 \pm 0.02$, compared with theoretical static/total pressure ratios of 0.265 for frozen composition and 0.29 for shifting equilibrium composition.

5. The combustor operated with a stable combustion chamber pressure; fluctuations were nominally ±5 psia. Stability was not significantly affected by the combustor O/F ratio, chamber pressure level, or percent seed flow.

6. The combustor has proved to be highly reliable and essentially free of operational difficulties. Run durations are limited only by the propellant storage capacity.

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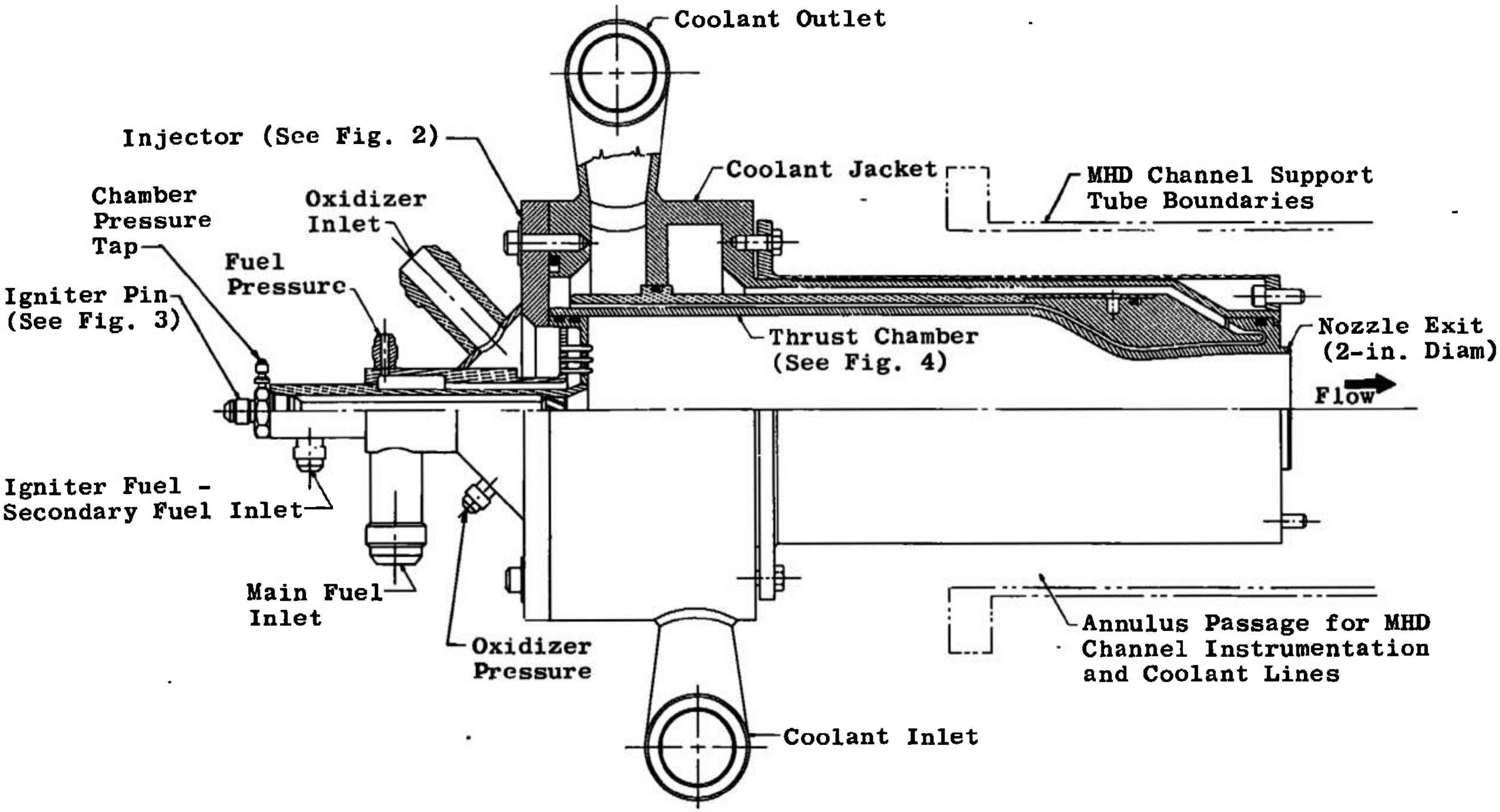
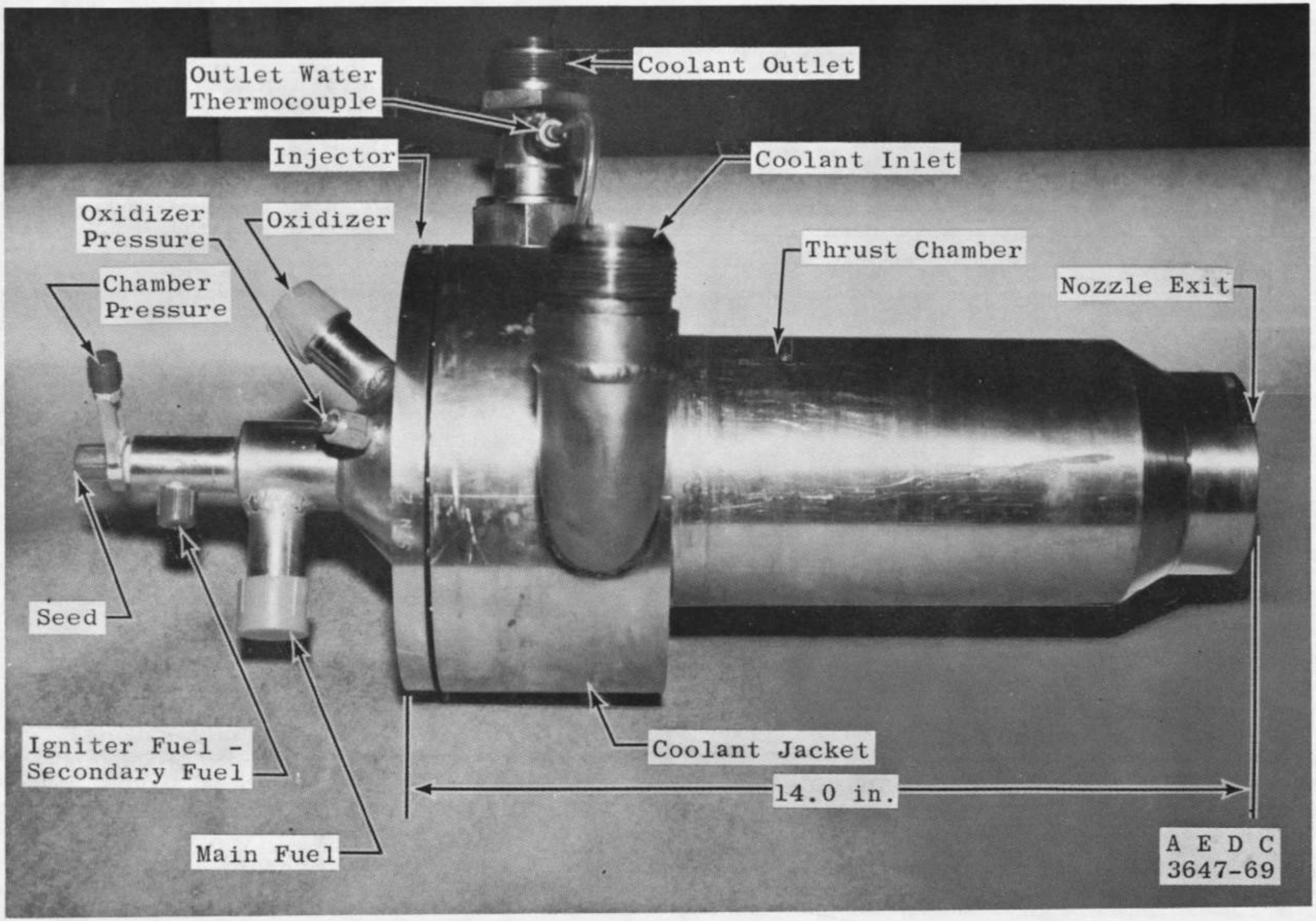
## APPENDIXES

I. ILLUSTRATIONS

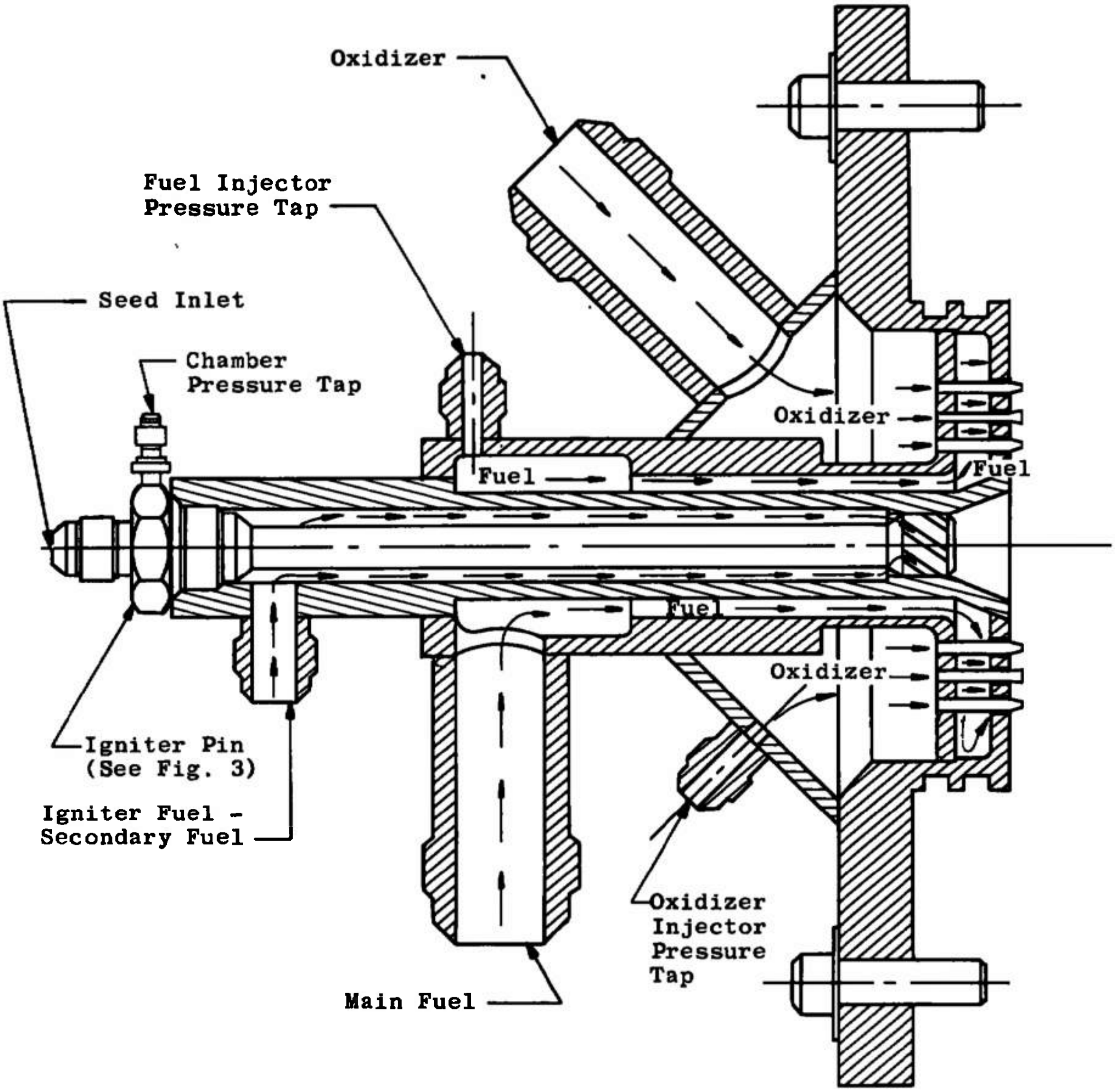
II. TABLES



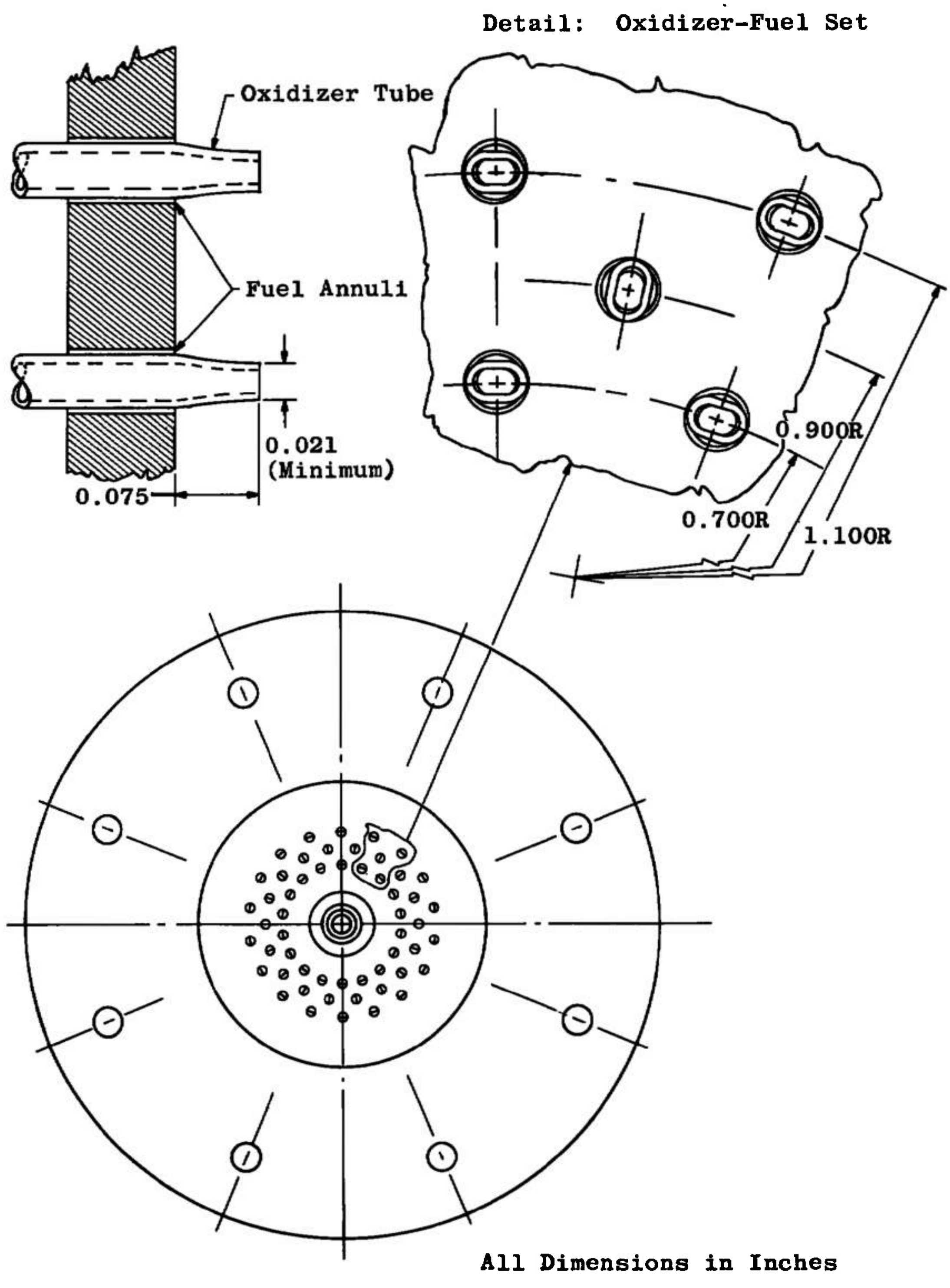

a. Schematic
Fig. 1 Combustor Assembly



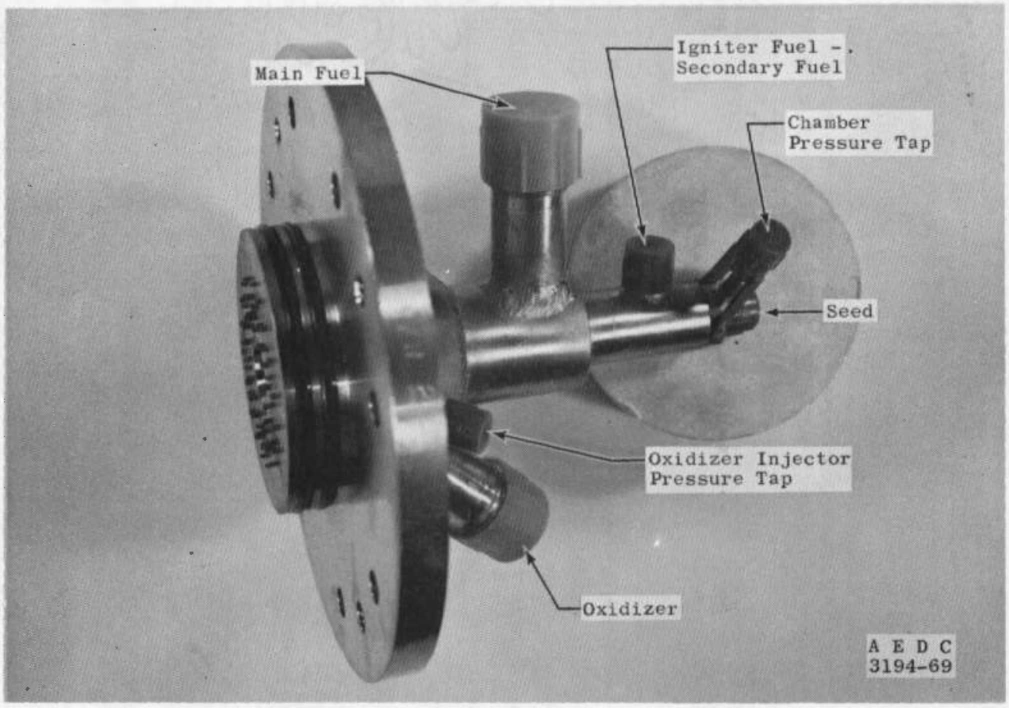

b. Photograph
Fig. 1 Concluded



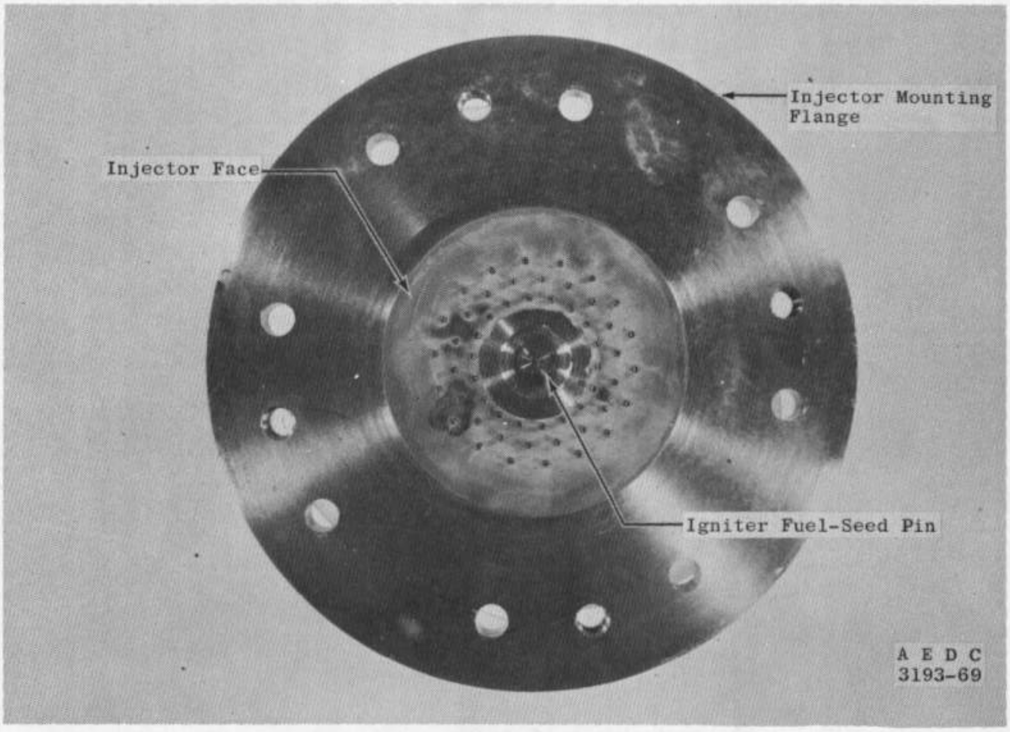

a. Schematic
Fig. 2 Injector



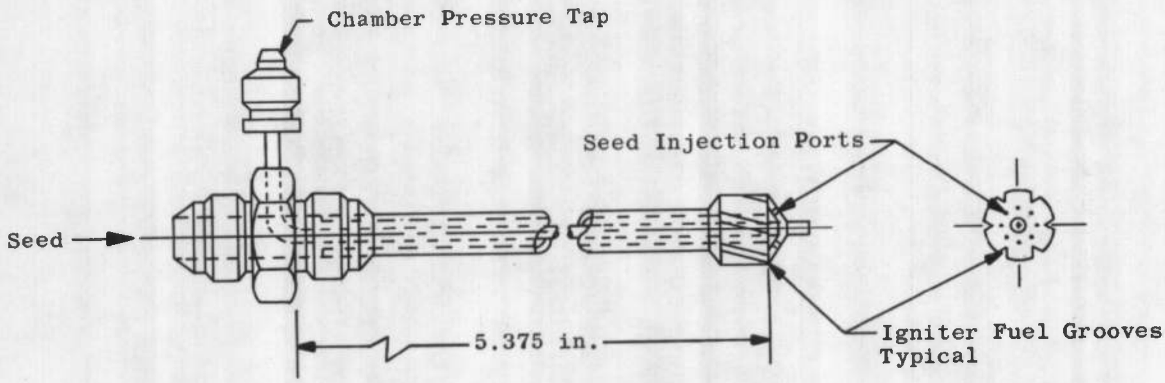

b. Injector Face Detail, Schematic
Fig. 2 Continued



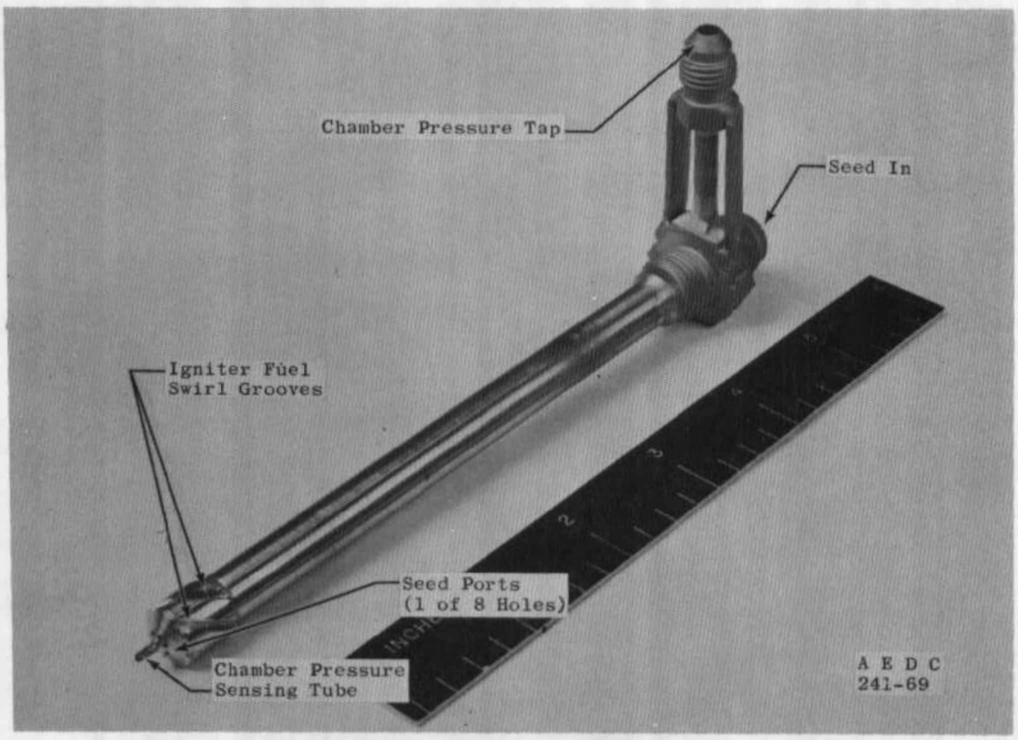

c. Injector Photograph
Fig. 2 Continued




d. Injector Face Photograph
Fig. 2 Concluded

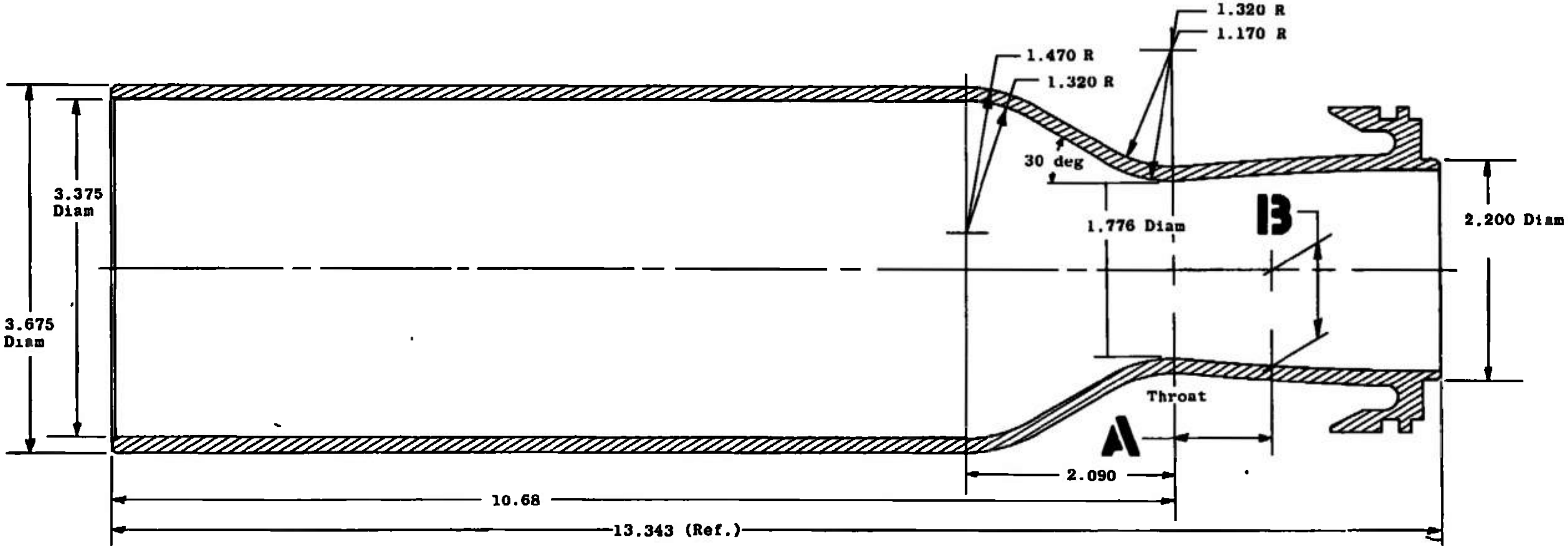



a. Schematic




b. Photograph

Fig. 3 Igniter Fuel-Seed Pin



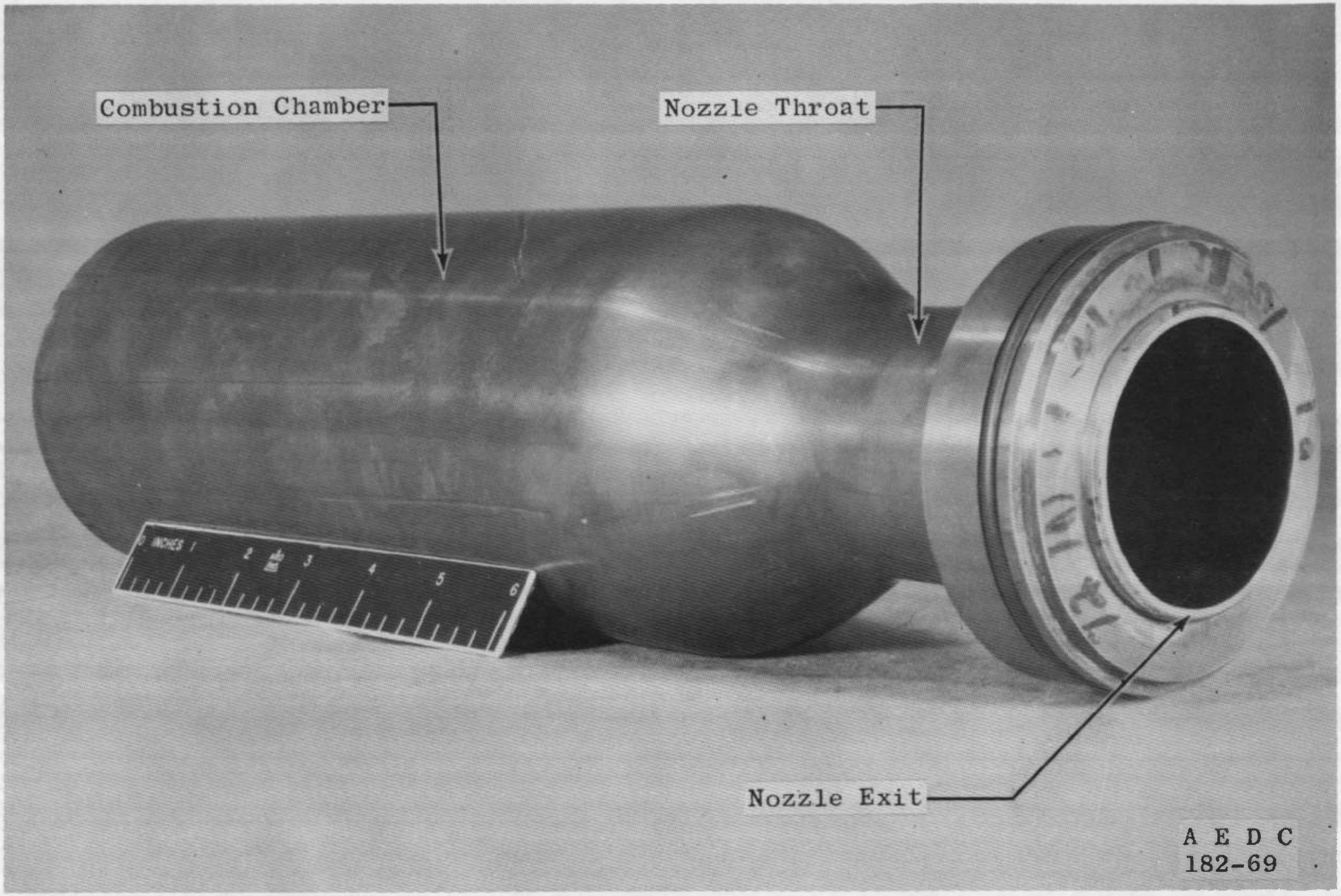

All Dimensions in Inches

| A | B | A | B |
|---|---|---|---|
| 0.000 | 0.888 | 1.350 | 0.962 |
| 0.050 | 0.888 | 1.450 | 0.967 |
| 0.150 | 0.888 | 1.550 | 0.973 |
| 0.250 | 0.889 | 1.650 | 0.978 |
| 0.350 | 0.892 | 1.750 | 0.982 |
| 0.450 | 0.897 | 1.850 | 0.986 |
| 0.550 | 0.904 | 1.950 | 0.990 |
| 0.650 | 0.911 | 2.050 | 0.992 |
| 0.750 | 0.918 | 2.150 | 0.995 |
| 0.850 | 0.926 | 2.250 | 0.997 |
| 0.950 | 0.934 | 2.350 | 0.998 |
| 1.050 | 0.941 | 2.450 | 0.999 |
| 1.150 | 0.948 | 2.550 | 1.000 |
| 1.250 | 0.955 | 2.663 | 1.000 |

Coordinates A and B Given to Nearest 0.001 in.

a. Schematic

Fig. 4 Thrust Chamber




b. Photograph
Fig. 4 Concluded

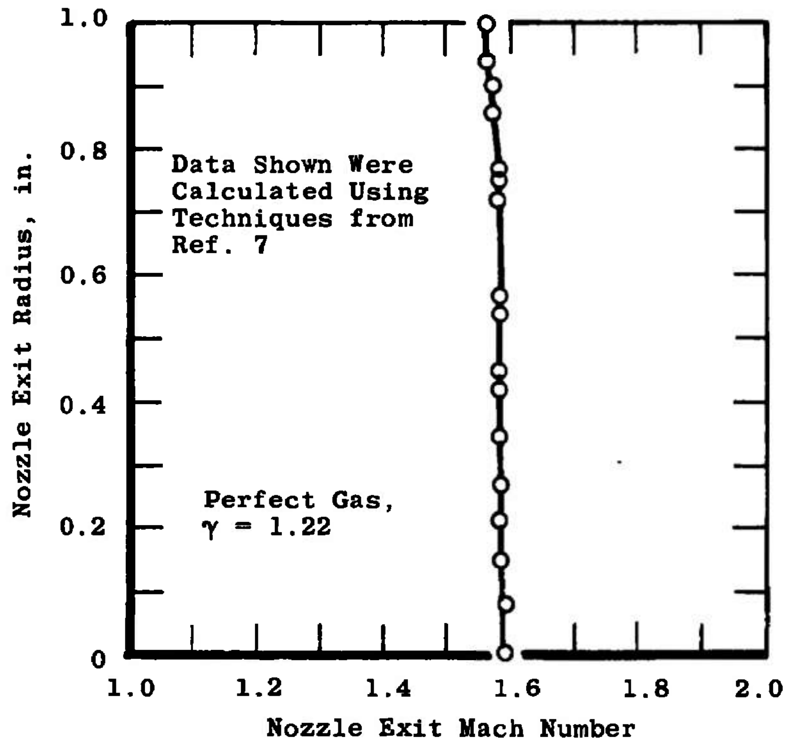



Fig. 5 Calculated Nozzle Exit Mach Number Profile

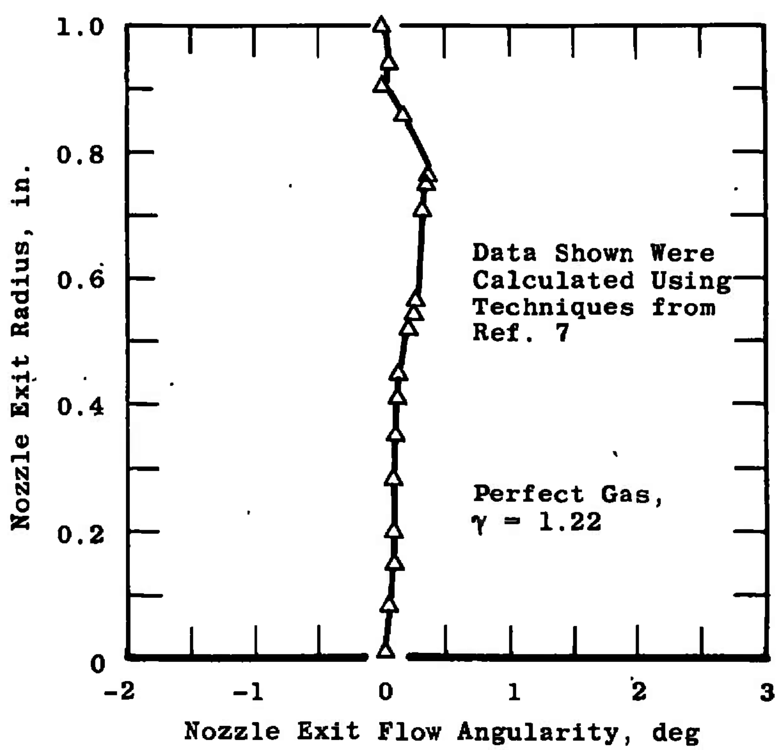
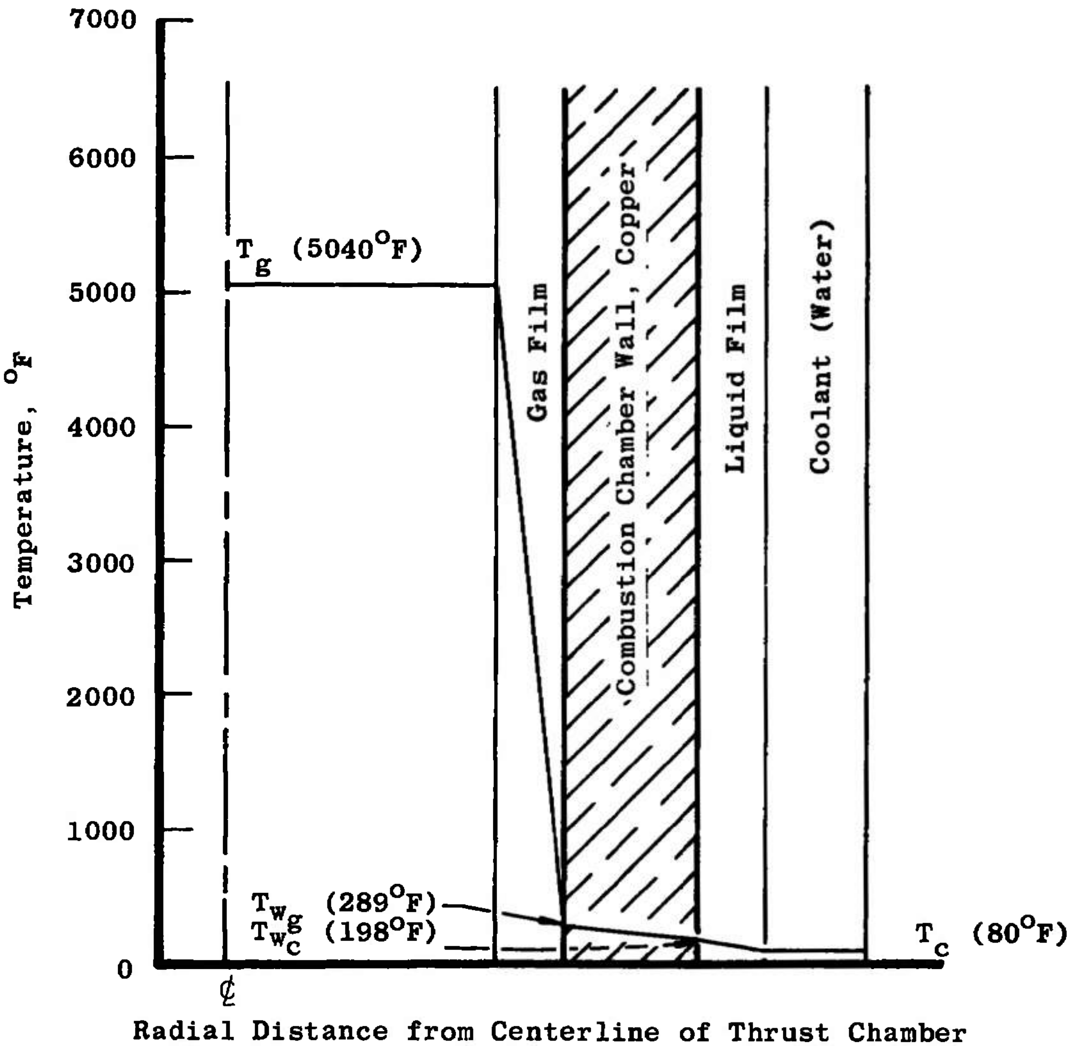



Fig. 6 Calculated Nozzle Exit Flow Angularity




Gas Film Coefficient, $H_g$ = 0.00055 Btu/sec/in.$^2$ °F

Liquid Film Coefficient, $H_\ell$ = 0.022 Btu/sec/in.$^2$ °F

Heat Transfer Rate, q = 2.61 Btu/sec/in.$^2$

Combustion Gas Static Temperature, $T_g$ = 5040°F

Chamber Gas Wall Temperature, $T_{w_g}$ = 289°F

Chamber Coolant Wall Temperature, $T_{w_c}$ = 198°F

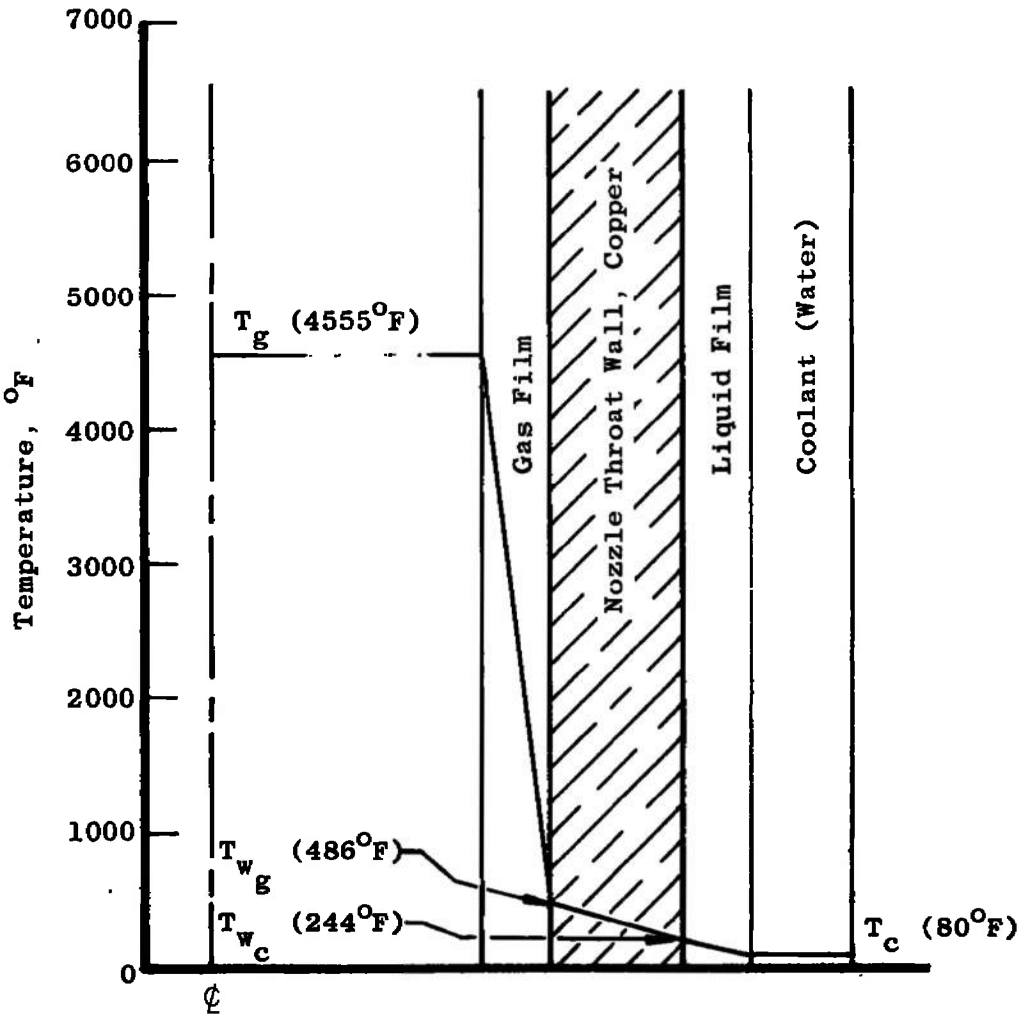
Coolant Temperature, $T_c$ = 80°F

Coolant Flow Rate, $\dot{w}_c$ = 15 $lb_m$/sec

Coolant Velocity, $V_c$ = 45 ft/sec

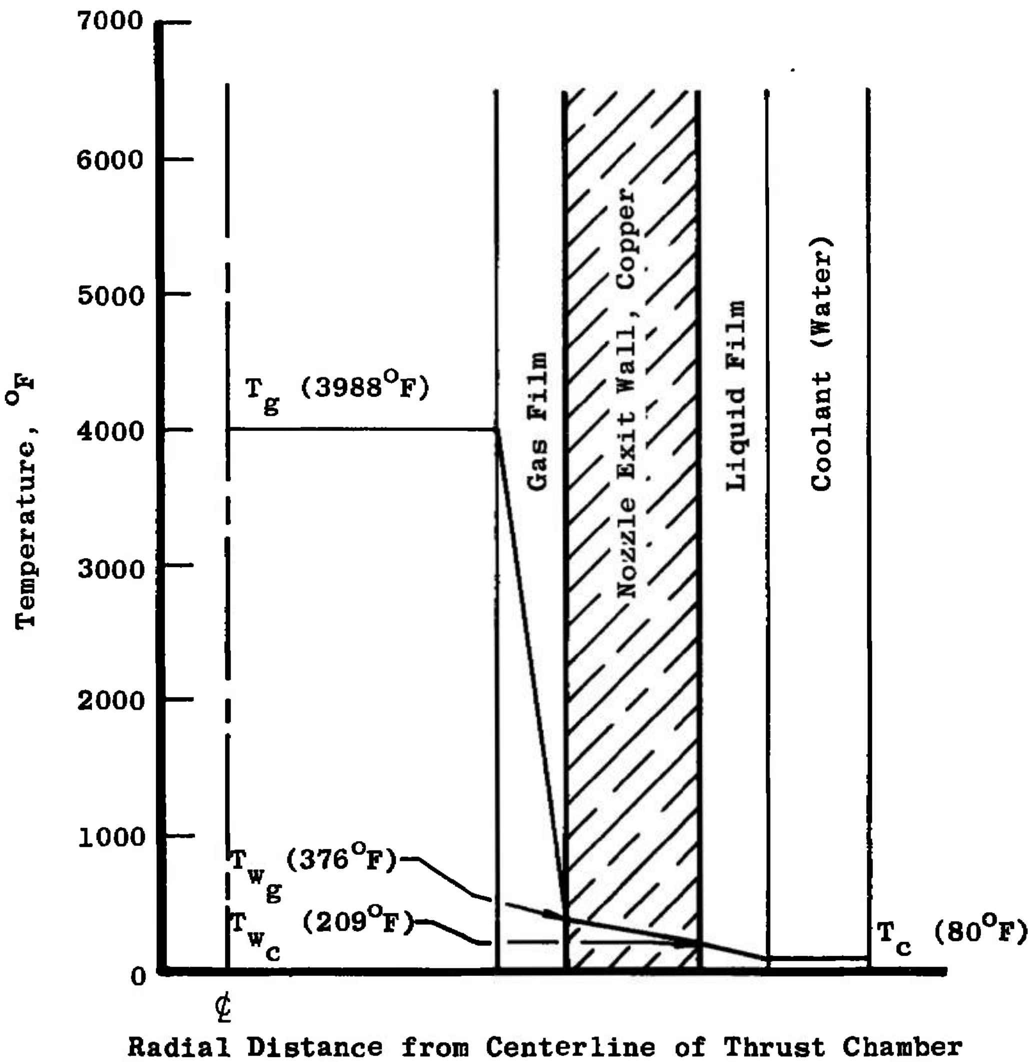
a. Combustion Chamber

Fig. 7 Typical Calculated Heat Transfer Results




Gas Film Coefficient, $H_g$ = 0.0017 Btu/sec/in.$^2$ °F
Liquid Film Coefficient, $H_\ell$ = 0.042 Btu/sec/in.$^2$ °F
Heat Transfer Rate, q = 6.93 Btu/sec/in.$^2$
Combustion Gas Static Temperature, $T_g$ = 4555°F
Chamber Gas Wall Temperature, $T_{w_g}$ = 486°F
Chamber Coolant Wall Temperature, $T_{w_c}$ = 244°F
Coolant Temperature, $T_c$ = 80°F
Coolant Flow Rate, $\dot{w}_c$ = 15 $lb_m$/sec
Coolant Velocity, $V_c$ = 90 ft/sec

b. Nozzle Throat
Fig. 7 Continued




Gas Film Coefficient, $H_g$ = 0.0013 Btu/sec/in.$^2$ °F

Liquid Film Coefficient, $H_\ell$ = 0.035 Btu/sec/in.$^2$ °F

Heat Transfer Rate, q = 4.79 Btu/sec/in.$^2$

Combustion Gas Static Temperature, $T_g$ = 3988°F

Chamber Gas Wall Temperature, $T_{w_g}$ = 376°F

Chamber Coolant Wall Temperature, $T_{w_c}$ = 209°F

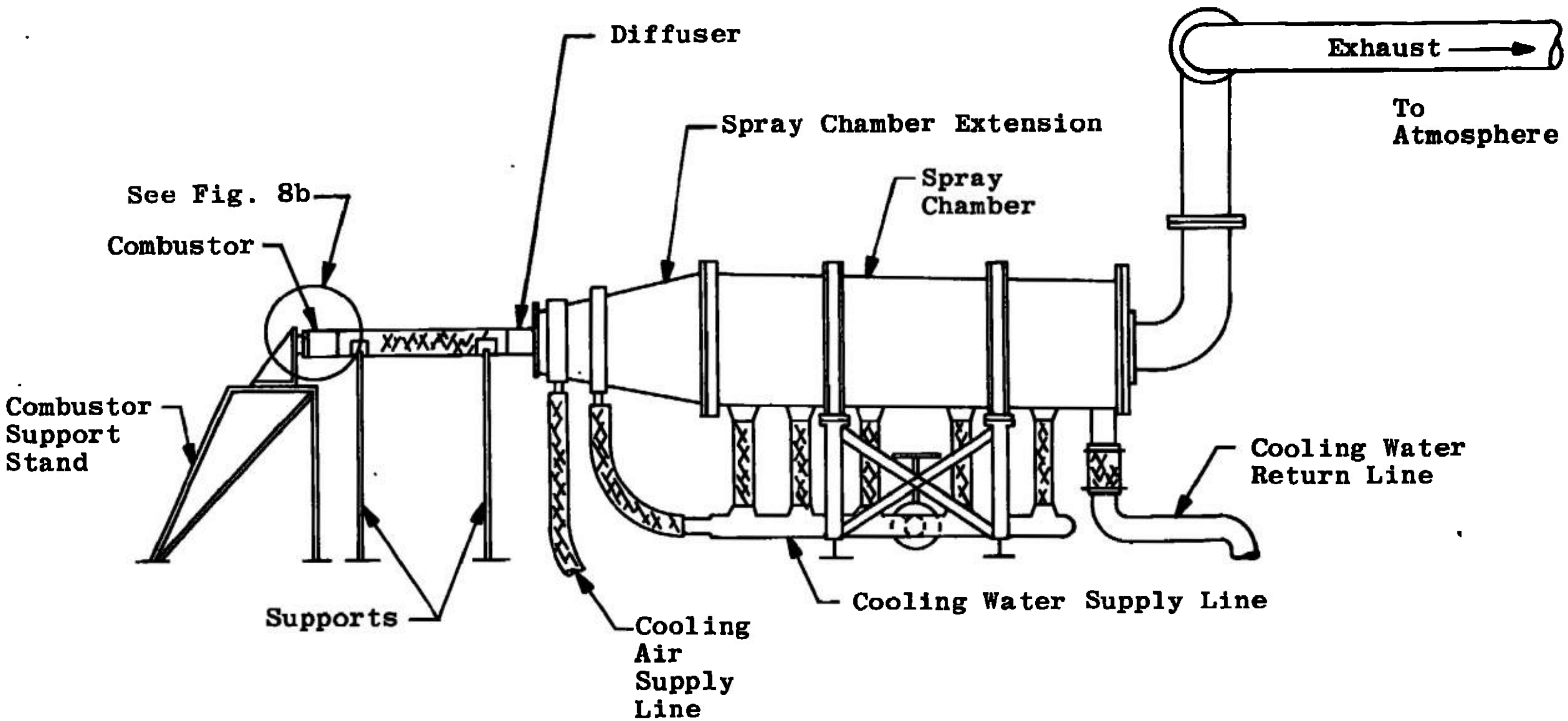
Coolant Temperature, $T_c$ = 80°F

Coolant Flow Rate, $\dot{w}_c$ = 15 $lb_m$/sec

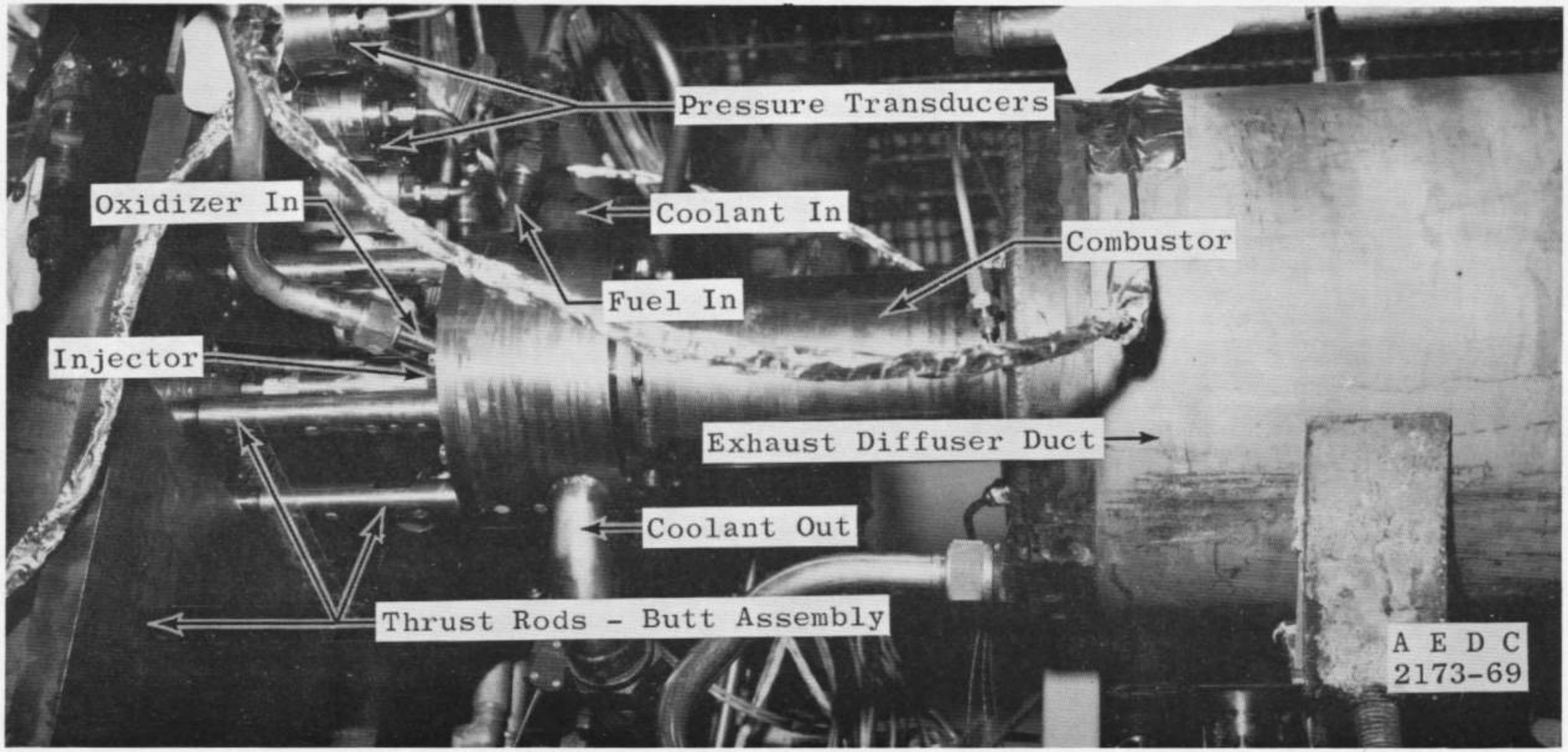
Coolant Velocity, $V_c$ = 80 ft/sec

c. Nozzle Exit

Fig. 7 Concluded

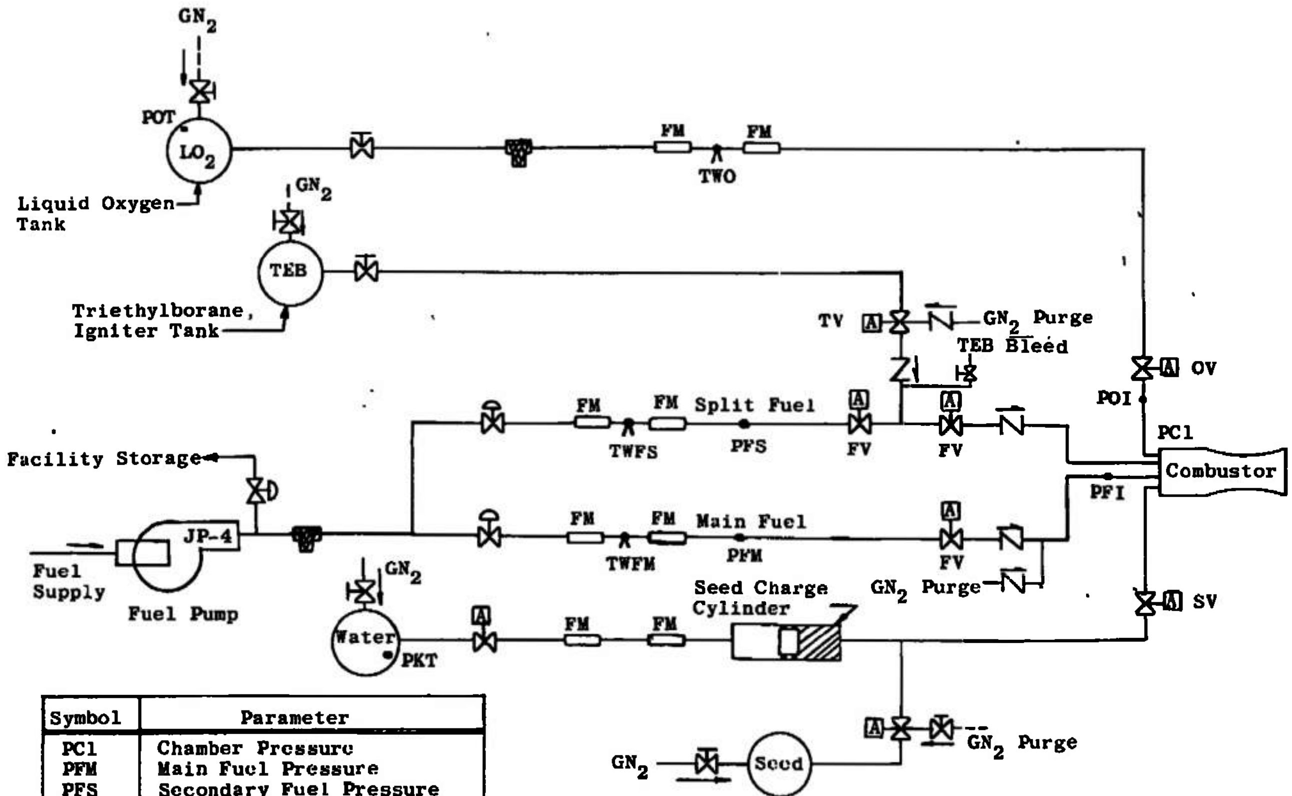



a. Schematic
Fig. 8 Combustor Installation




b. Photograph

Fig. 8 Concluded




| Symbol | Parameter |
|---|---|
| PC1 | Chamber Pressure |
| PFM | Main Fuel Pressure |
| PFS | Secondary Fuel Pressure |
| POT | $LO_2$ Tank Pressure |
| PKT | Seed Tank Pressure |
| POI | $LO_2$ Injector Pressure |
| PFI | Fuel Injector Pressure |
| TWFM | Main Fuel Temperature |
| TWFS | Secondary Fuel Temperature |
| TWO | $LO_2$ Temperature |
| FV | Fuel Valve |
| OV | $LO_2$ Valve |
| TV | TEB Valve |
| SV | Seed Valve |
| FM | Flowmeter |
| $GN_2$ | Gaseous Nitrogen |

Fig. 9 Propellant and Seed Flow Schematic

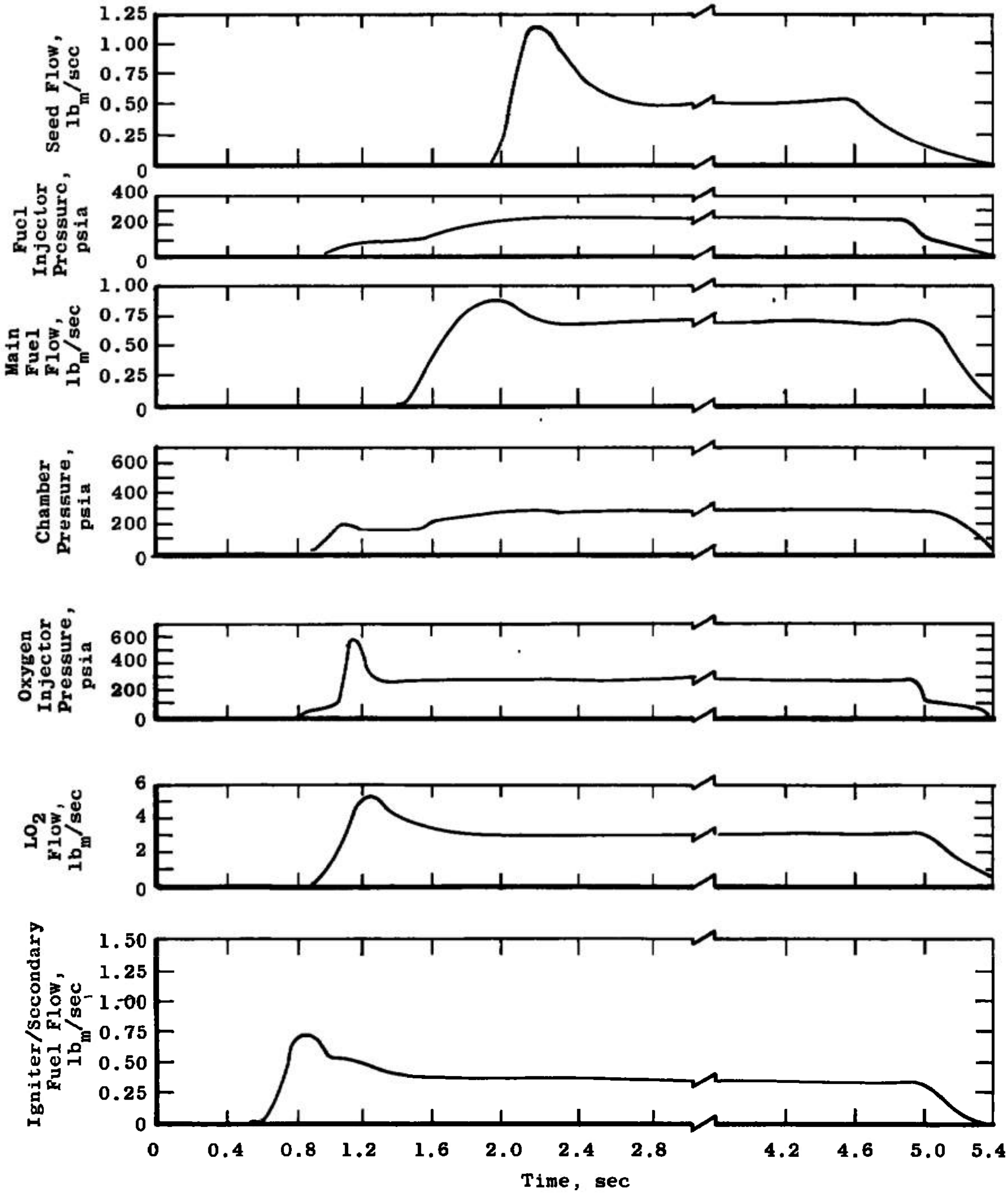



Fig. 10 Typical Combustor Firing Sequence

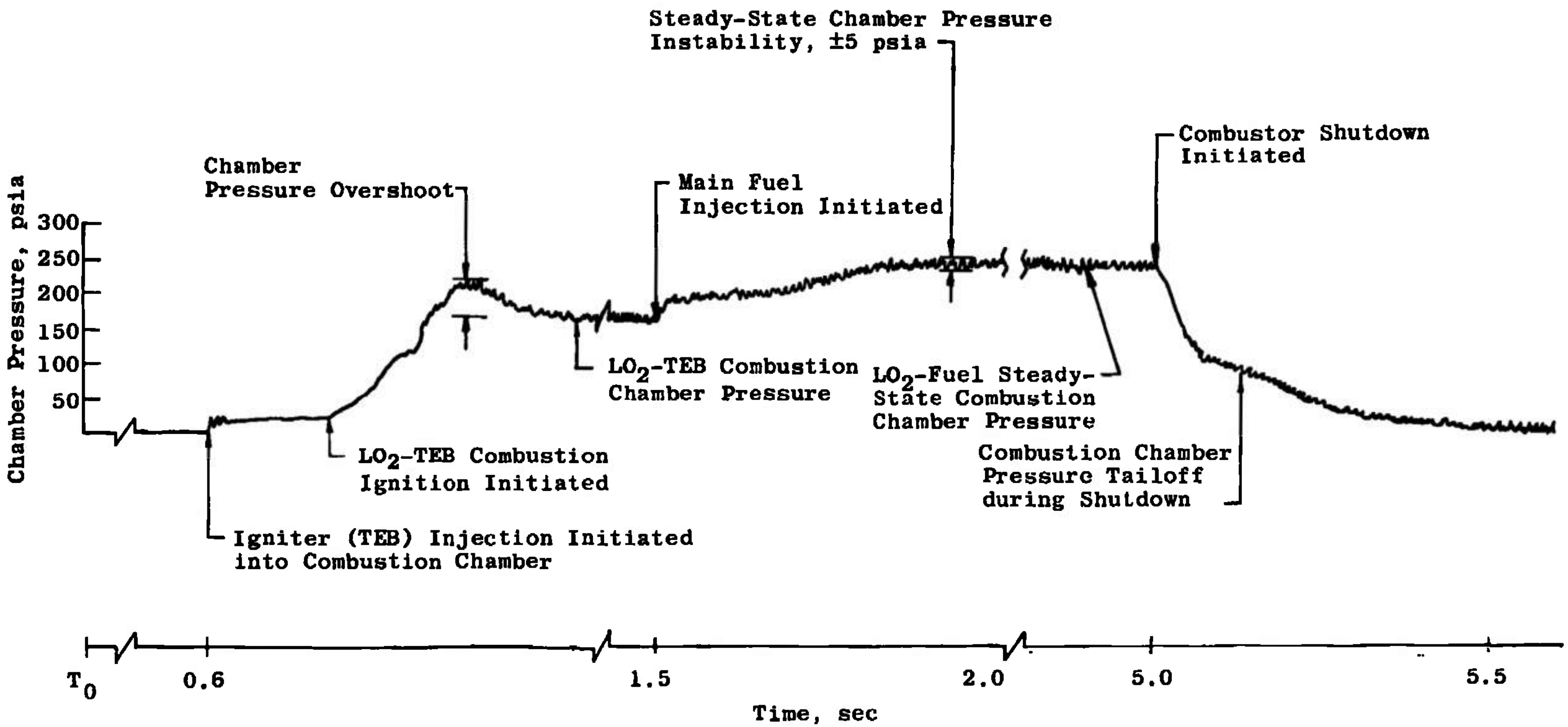
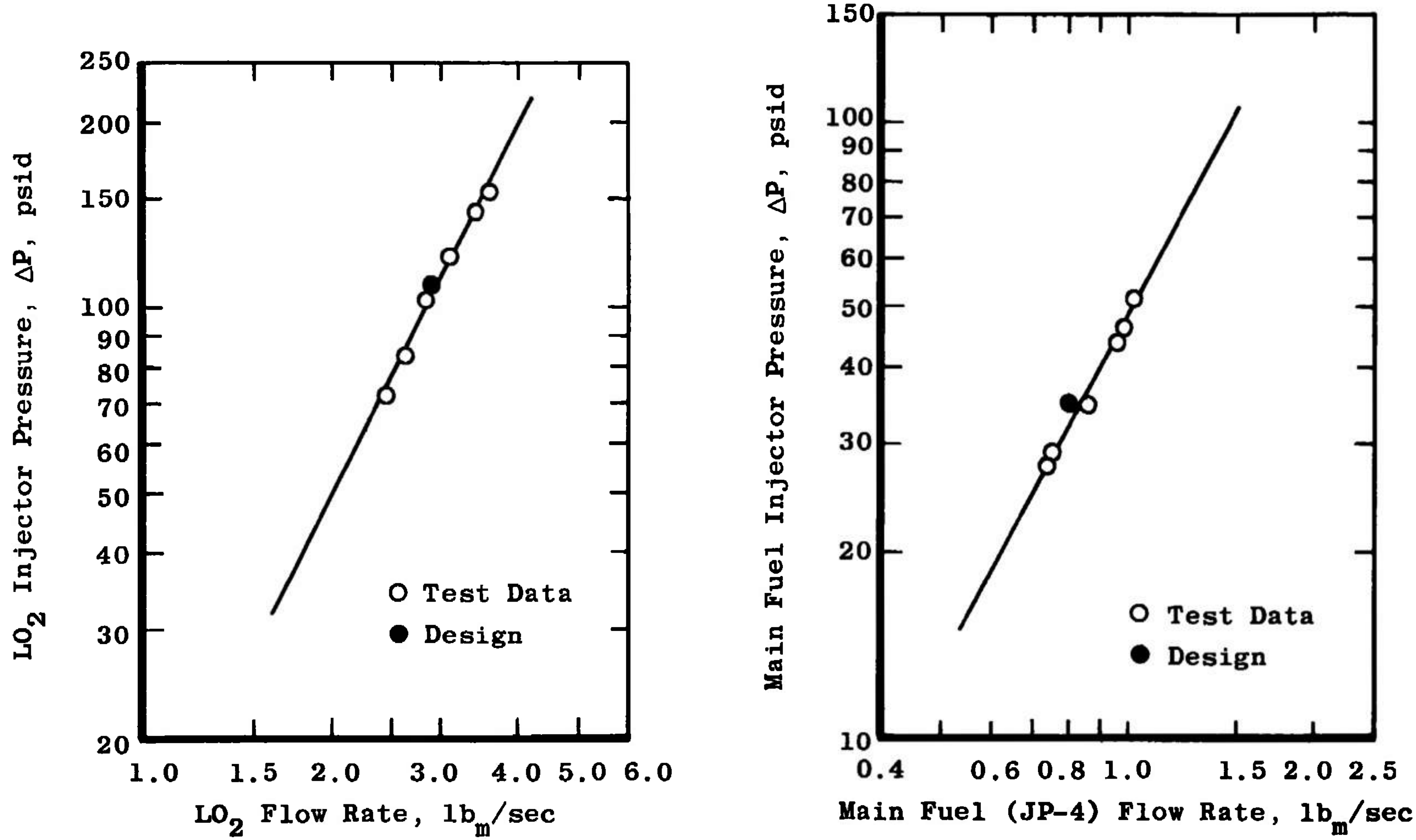


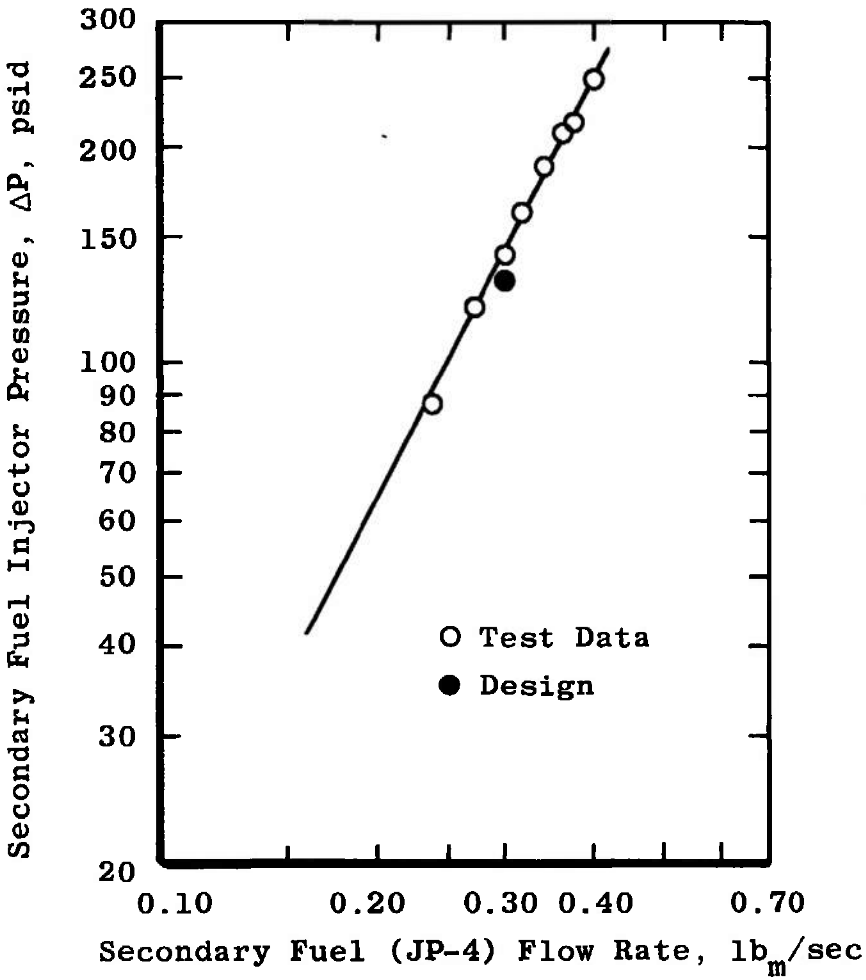

Fig. 11 Typical Combustion Chamber Pressure Data (without Seed)



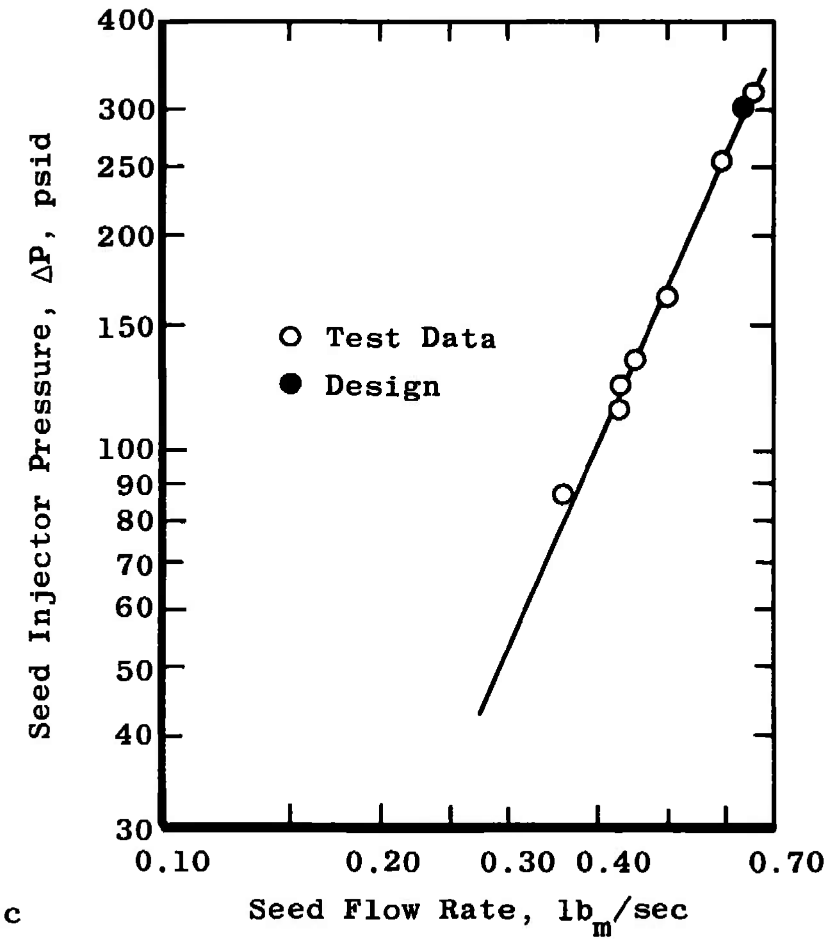

a. Liquid Oxygen

b. Main Fuel (JP-4)

Fig. 12 Injector Pressure Drop Variation with Flow Rate




c. Secondary Fuel (JP-4)




d. Seed

Fig. 12 Concluded

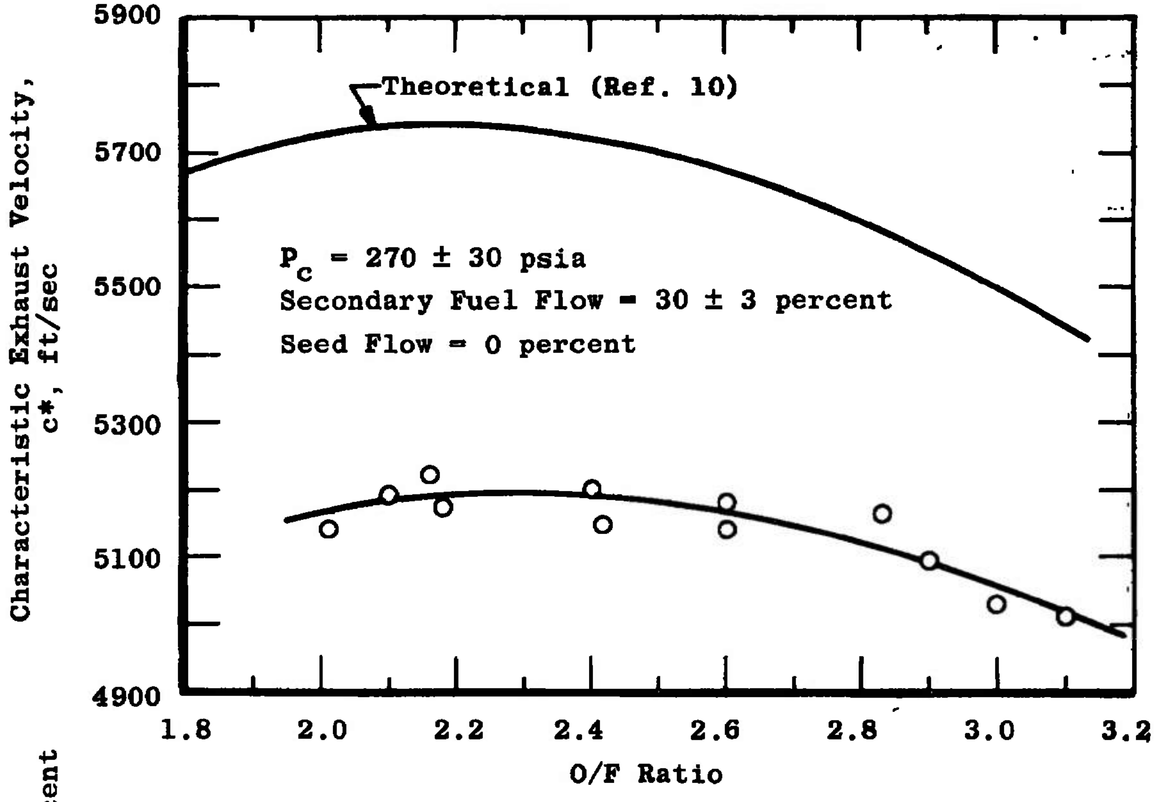



Fig. 13 Combustor Characteristic Exhaust Velocity Variation with Oxidizer-to-Fuel Ratio

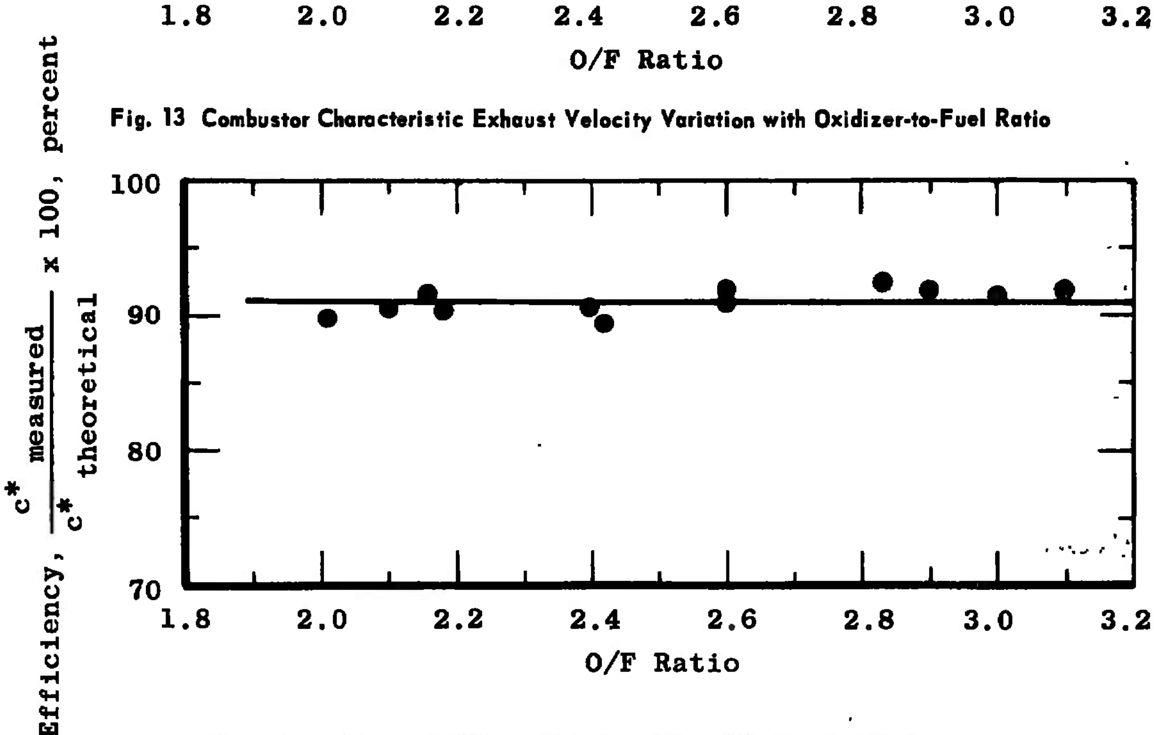



Fig. 14 Combustor Efficiency Variation with Oxidizer-to-Fuel Ratio

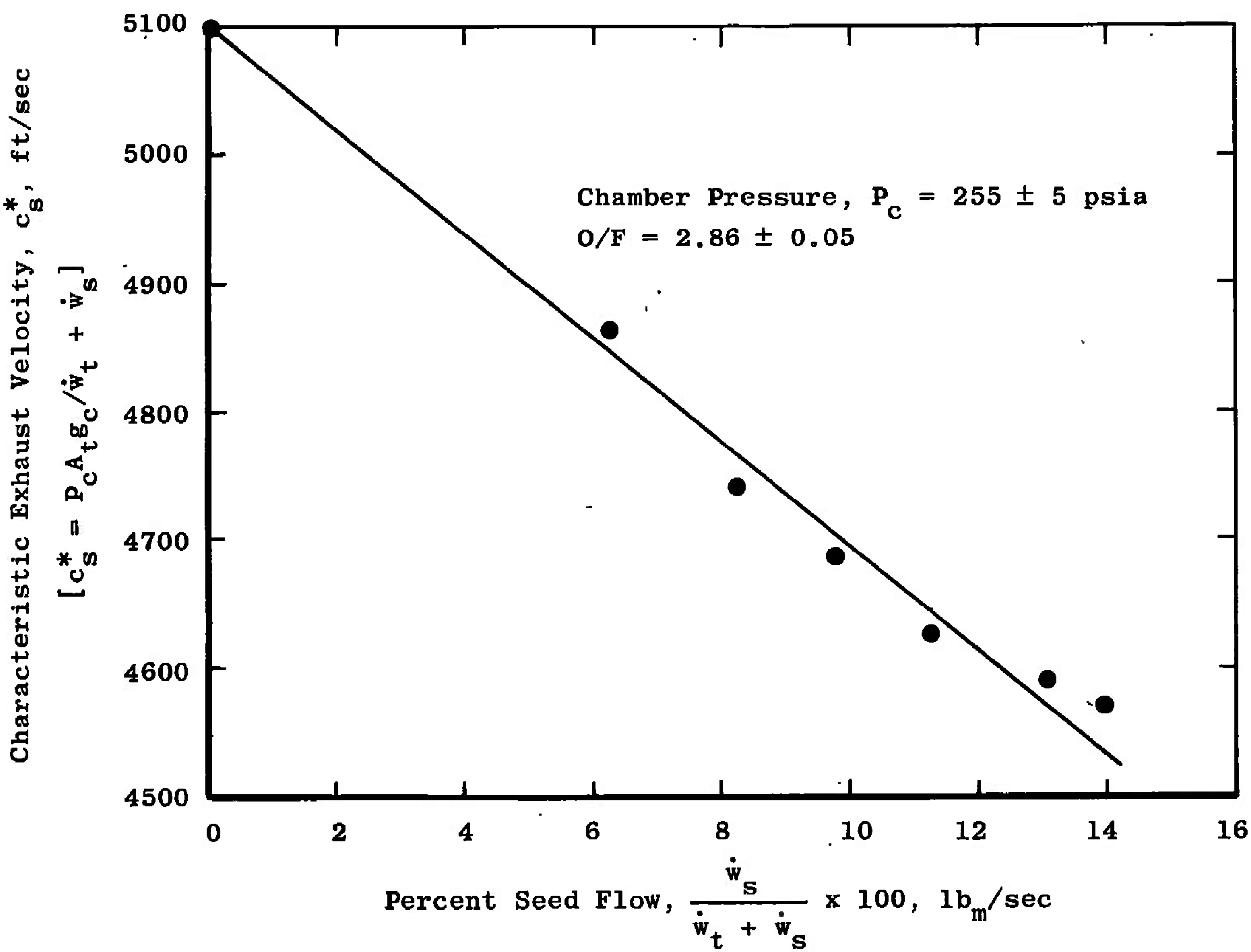
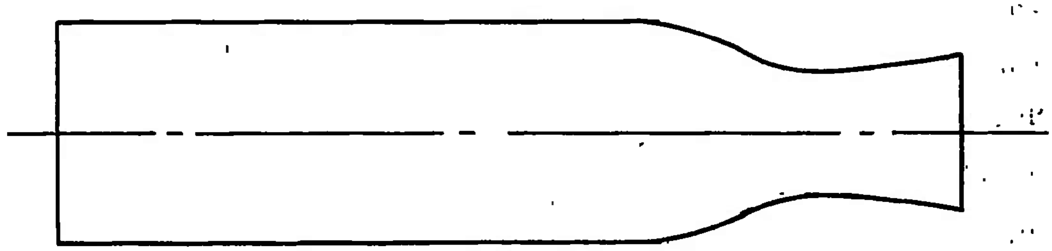



Fig. 15 Effect of Percent Seed Flow Rate on Combustor Characteristic Exhaust Velocity



Thrust Chamber Contour

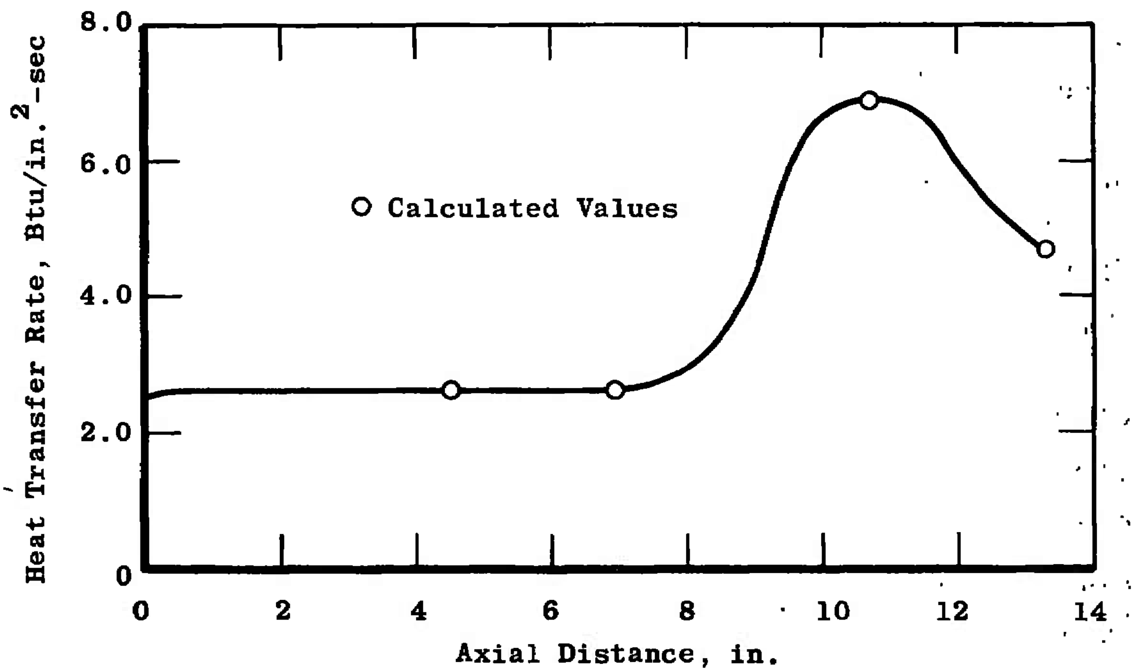



Fig. 16 Calculated Heat Transfer Rates Extrapolated along the Axial Length of the Thrust Chamber Wall

**TABLE I**
**DESIGN DATA FOR THE COMBUSTOR**

| | |
|---|---|
| Nozzle Shape | Contoured |
| Nozzle Exit Half-Angle | 0 deg |
| Nozzle Throat Diameter | 1.776 in. |
| Nozzle Exit Diameter | 2.00 in. |
| Nozzle Area Ratio | 1.27 |
| Injector Type | Coaxial |
| Injector Material | 347 Stainless Steel |
| Injector Oxidizer Velocity at O/F = 2.6 | 95 ft/sec |
| Injector Fuel Velocity at O/F = 2.6 | 45 ft/sec |
| Injector Oxidizer Pressure Drop at O/F = 2.6 | 110 psid |
| Injector Fuel Pressure Drop at O/F = 2.6 | 35 psid |
| Thrust Chamber Material | Mallory 3 Copper Alloy |
| Combustion Chamber Length (Injector Face to Throat) | 10 in. |
| Characteristic Length, L* | 33 in. |
| Chamber Contraction Ratio | 3.61 |
| Combustor Coolant (Water) Flow | 15.0 $lb_m$/sec |
| Combustor Coolant Velocity, Throat | 90 ft/sec |
| Combustor Coolant Velocity, Chamber | 45 ft/sec |
| Propellant Total Flow Rate | 4.0 $lb_m$/sec |
| Oxidizer Flow | 2.9 $lb_m$/sec |
| Main Fuel Flow | 0.8 $lb_m$/sec |
| Secondary Fuel Flow | 0.3 $lb_m$/sec |
| O/F Ratio | 2.6 |
| Chamber Pressure | 270 psia |

## TABLE II
## INSTRUMENTATION

| Parameter Designation | Estimated Measurement Uncertainty (2 Sigma) Steady-State: Percent of Reading | Estimated Measurement Uncertainty (2 Sigma) Steady-State: Units of Measurement | Range of Measurement | Measuring Device | Recording Device | Method of System Calibration |
|---|---|---|---|---|---|---|
| Combustion Chamber Pressure | ±0.5 | | 125 to 300 psia | Bonded Strain-Gage Pressure Transducers | Sequential Sampling, Millivolt-to-Digital Converter and Magnetic Tape Storage Data Acquisition System | Resistance Shunt Based on the Standards Laboratory Determination of Transducer Applied Pressure versus Resistance Shunt Equivalent Pressure Relationship |
| Propellant Tank Pressure ($LO_2$) | ±0.5 | | 200 to 600 psia | | | |
| Seed Tank Pressure | ±0.5 | | 250 to 700 psia | | | |
| Fuel Regulator Pressure | ±0.5 | | 200 to 600 psia | | | |
| Injector Pressures ($LO_2$, JP-4, TEB Seed) | ±0.5 | | 200 to 550 psia | | | |
| Nozzle Exit Static Pressures | | ±0.375 psi | 40 to 75 psia | | | |
| | ±0.5 | | 75 to 100 psia | | | |
| Propellant and Cooling Water Temperatures | | ±(1.67°F + 0.67% reading) | -300 to -250°F | Copper-Constantan Temperature Transducers | | Millivolt Substitution Based on the NBS Temperature versus Millivolt Tables |
| | | ±3.0°F | 50 to 90°F | Iron-Constantan Temperature Transducers | | |
| $LO_2$ Flow | ±2.33 | | 2 to 3.5 $lb_m$/sec | Turbine Volumetric Flow Transducers | Frequency-to-Voltage Converter onto Sequential Sampling, Millivolt-to-Digital Converter and Magnetic Tape Storage Data Acquisition System | Frequency Substitution Based on the Standards Laboratory Determination of Transducer Water Volumetric Flow versus Frequency Output Relationship |
| JP-4 Main Flow | ±1.22 | | 0.6 to 1.4 $lb_m$/sec | | | |
| Seed Flow | ±0.934 | | 0.06 to 0.6 $lb_m$/sec | | | |
| JP-4 Secondary Flow | ±0.961 | | 0.05 to 0.5 $lb_m$/sec | | | |
| Coolant Flow | ±0.5 | | 13 to 16 $lb_m$/sec | | | |

UNCLASSIFIED

Security Classification

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13. ABSTRACT


Design characteristics and performance of a combustor for use as a high energy, ionized gas source in magnetohydrodynamic power generator studies are described. The liquid oxygen ($LO_2$)/JP-4 combustor was operated over a chamber pressure range from 240 to 300 psia and at a characteristic exhaust velocity efficiency of 91 ± 1 percent for oxidizer/fuel ratios ranging from 2.0 to 3.1. Combustor power output was approximately 17.5 to 20.5 megawatts (MW) over its range of operation. Provisions were incorporated into the design for injection of a saturated solution of water and cesium carbonate seeding agent into the thrust chamber to provide a high ion concentration in the exhaust gases.



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| 14. KEY WORDS | LINK A | | LINK B | | LINK C | |
|---|---|---|---|---|---|---|
| | ROLE | WT | ROLE | WT | ROLE | WT |
| combustion chambers | | | | | | |
| design | | | | | | |
| characteristics | | | | | | |
| performance | | | | | | |
| magnetohydrodynamic generators | | | | | | |
| 2 ionized gases | | | | | | |
| injectors 3 | | | | | | |
| 1 MHD generators | | | | | | |
| 15-20 | | | | | | |

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